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DETAILED PRESIMULATION REPORT FOR ABORT SIMULATION VI [U]

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ENG. 345

GRUMMAN AIRCRAFT ENGINEERING CORPORATION

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PART I

GENERAL INFORMATION

1. INTRODUCTION

The Abort Simulation Program is a manned simulation study with the full six degrees of freedom. The program is run in real time and simulates aborts from the powered descent portion of the LEM trajectory. These aborts occur at pre-selected points along the powered descent. Five points along a descent trajectory have been chosen.

A moving base motion device driven by computer signals provides for three axes rotational motion in response to pilot inputs. Three projectors provide the pilot with a starfield, lunar scene, horizon and CSM position. An instrument panel mounted in the cockpit has TV displays, a three axes attitude ball and synchro driven meters. Displayed information will be evaluated to determine the optimum information required for an abort mission. A control console located in the computer room contains repeater instruments which follow the cockpit displays.

There are two ASI-210 digital computers programmed in fixed point providing a real time simulation to the vehicle equations of motion. There are 34 digital-to-analog (D-A) converter channels in use. Time sharing of these channels allows for 56 D-A conversions. Six analog-to-digital (A-D) conversions are used. Mechanization of the control system, throttle outputs, drives for displays, recorders, projectors, and moving base requires about 210 analog operational amplifiers.

2. CONTROL SYSTEM

The attitude control system is a three axis system which performs an analog integration of the digitally computed angular accelerations, $\dot{p}$, $\dot{q}$ and $\dot{r}$. The operating modes for rotational and translational control are manually selected by the pilot. For manual control the pilot will select either the Attitude Hold or Rate Command Mode. In the Attitude Hold Mode there is attitude feedback (Sp, Sq, Sr) in all three axes when the input to the control system from the rotational controller is zero. The output from the rotational controller passes through a deadband which limits the input to zero for 10% of the maximum controller output. This is to compensate for mechanical slop in the controller. Once the controller output passes through this deadband the control system functions as it would in the Rate Command Mode. For error signals greater than 0.8 degree, a four jet rotational couple is produced. A two jet couple is produced for error signals less than .8°.

There are three pilot operated switches for emergency mode of operation. They are selected separately for each axis. In the emergency mode, the modulator produces a pulse train or a direct on-off signal. Direct on-off mode is initiated when the rotational controller exceeds 75% of the controller throw.

2. CONTROL SYSTEM (Cont'd)

There are two translational control modes, Pulse Train Modulation (PTM) and Direct on-off. They are selected by a switch located on the instrument panel. Four hundred pounds of thrust along the X axis may be selected by throwing a switch located on the throttle controller. A complete description of pilot operated switches appears in Part II.

3. MODULATORS

Pulse modulators receive error signals from the Attitude Control System and the jet select logic determines which pair of the 16 Reaction Control System (RCS) jet engines should fire for rotational and/or translational control. The modulators and jet select logic are physically housed in what is referred to as a "Logic Box". The output of the Logic Box is interfaced with the digital computer. The Pulse Ratio Modulator produces a signal whose width and repetition frequency are dependent upon the normalized input error signal. There are three such modulators, one for each control channel, which provide rotation torque commands to the jet select logic. Pulse train modulators generate a signal which is fixed in amplitude, width and frequency. The equations representing the jet-select logic are in the Appendix.

4. ABORT PROCEDURE

The pilot will assume manual control when the abort light in the cockpit goes on. The light will go on at pre-determined points along the descent trajectory. Prior to this, the vehicle is in an "Automatic Mode." In order to initiate manual control the pilot must move the Attitude Mode Select switch out of "Auto" into either an "Attitude Hold" or "Rate Command Mode," both of which are manual. Upon manual take-over, the pilot will immediately start erecting the LEM at a 10°/sec rate. The appropriate attitude display must then be selected by pushing the Ascent Display Selector switch. Depending on which engine configuration is being used, the pilot may stage the descent engine and fire the ascent engine or continue to utilize the descent engine. An inertial or line-of-sight pitch program will then be followed. The pitch program shall be determined prior to the start of the run. At the completion of this pitch program, the LEM will nominally be in a circular parking orbit. Insertion into a transfer orbit will be effected after remaining in the parking orbit for a period of time. This dwell time will range from two to twenty minutes depending on the phasing of the LEM and CSM.

The procedure to be followed in the inertial pitch program is to rotate the LEM at 10°/sec until a zero pitch angle (erect attitude) is achieved. This angle is measured with respect to the landing site local vertical. The angle is held until a specified value of ΔV USED is displayed at which time the vehicle is rotated to a specified pitch angle at 10°/sec and held at that angle until another specified value of ΔV USED is achieved. This process is repeated until ΔV USED required for burnout is achieved. At the completion of this maneuver, the vehicle should be in a circular parking orbit.

4. ABORT PROCEDURE (Cont'd)

Nominally, no mid-course corrections are required since the LEM will be on a collision course with the CSM. In practice, it will be necessary for the pilot to make corrections as required due to manual errors induced. Except when mid-course corrections are being made, the vehicle shall be coasting. A run will be terminated at a line-of-sight (LOS) range of 30 nautical miles which is the initial condition for start of rendezvous.

In order to reduce the run time, a jump-time capability has been built into the simulator. The pilot will call for a jump in time during the coasting phase. The experimenter at the control console will determine the amount of time to be jumped from calculated data. He will then initiate the jump in time. After the final mid-course correction has been made, the run will be terminated by a jump to the 30 nautical mile range point. This jump time capability will result in an average saving of 40 minutes for a complete run.

An experimental test plan written by Crew Systems, appears in Part II. This test plan describes in detail, the experimental procedures and run schedule to be followed.

5. DIGITAL

Trajectory computations during the "Auto" descent phase will be an open loop computation implemented digitally by a process of continually up-dating initial conditions. The polynomial equations representing the descent guidance law during "Auto" descent are enclosed in the Appendix. The six degree of freedom equations are implemented digitally at an interation rate of 50 ms. There are a total of 34 D-A and 6 A-D channels in use for this simulation. Time sharing of the D-A converters expands the number of conversions to 56. The sampling and conversion rate depends on the program requirements with a minimum rate of 300 μ sec available. Access time to the internal memory is 2 μ sec. Addition and subtraction can be performed in 6 μ sec. The internal machine memory is 4,096, 21 bit words. A total of about 10,000 words are required for the simulation. This necessitated the use of an external tape for the "jump-time" routine. This routine is used only during the coasting phase and replaces the routine for the Auto Descent Trajectory which is read out of the machine after abort is initiated.

The equations of motion were programmed with a view to make maximum usage of standard sub-routines which have been developed and pre-tested in other programs. The routines are general and are written with speed and accuracy as the prime considerations. The simulation has been programmed in discrete blocks which allows for ease in effecting a change in any of the equations. An executive routine controls the up-dating and execution of the entire program.

The attitude engine thrusts are integrated digitally during the 50 ms iteration period and summed into the trajectory computation at synchronous points. Inertial attitude is solved digitally by integration of body axis angular rate into the direction cosine set. An excessive amount of analog equipment would be required for an analog implementation of attitude with a loss in accuracy. Digital implementation of mass and fuel use equations preserves the correlation between fuel use and trajectory perturbation. The anticipated accuracy of an unperturbed trajectory is expected to be $\pm$.0005ft/sec²

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6. ANALOG

Analog implementation was required for driving of the analog instruments, visual displays, strip chart recorders, x-y plotters and the moving base. The equations used to drive the moving base and projectors are in the Appendix. The attitude control system is an analog mechanization which receives pilot inputs from the throttle and controller which are also implemented on the analog computers.

7. DATA RECORDING

Analog time histories, x-y plots and digital printouts will be made of the parameters indicated on the enclosed chart. The analog time histories will provide a continuous recording of data. Any discontinuity which may occur between digital printouts would be apparent. Dynamic characteristics and frequency response of the vehicle may be evaluated from the analog data.

Digital printout of data will occur every 10 seconds when a main engine is thrusting, every 5 minutes when not, and at special events. These special events are tabulated on the enclosed chart. If a printout is called for while a printout is being made, the data will be stored in the computer memory and printed out at the completion of the previous printout. The system has the capability of back logging a total of four printouts. The line printer can output 200 lines per minute.

8. SPECIAL EVENT PRINTOUT

- Energizing of the Abort Light
- Manual Take-over
- Firing of the Descent or Ascent Engines
- Shut Down of the Descent or Ascent Engines
- Staging
- Energizing the "Freeze Button"
- Executing or stopping a Translational Command
- Initiation of a "Jump-Time"

9. DIGITAL AND ANALOG PRINTOUTS

DIGITAL PRINTOUT

V_{r1B}	ΔV_{TR}
V_{n1B}	ΔV_m
V_{n2B}	ΔV_A
ϵ_B	ΔV TO GO
λ_E	M_T
R_E	p
$\dot{\rho}$	q
V_j	r
V_k	$\theta,\ \theta_t,\ \theta_A$
V_{RT}	$\psi,\ \psi_t,\ \psi_A$
ρ	$\phi,\ \phi_t,\ \phi_A$
A	t
E	$\left[\int(\epsilon\ \theta,\ \phi,\ \psi)dt\right]^2$
R_{xA}	$\dot{\epsilon}_B$
R_{yA}	$\dot{\lambda}_E$
R_{zA}	$\partial\psi_m$
ΔM_{jx+}	$\partial\theta_m$
ΔM_{jx-}	ΔM_{jz+}
ΔM_{jy+}	ΔM_{jz-}
ΔM_{jy-}	

ANALOG TIME HISTORIES

$\dot{\rho}$	ϵ_p
V_j	$\dot{\epsilon}_q$
V_k	ϵ_r
ρ	ΔM_a
A	ΔM_B
E	M_T
ΔM_{jx+}	ΔV_{TR}
ΔM_{jx-}	R_B
ΔM_{jy+}	
ΔM_{jy-}	
ΔM_{jz+}	
ΔM_{jz-}	
θ	
ψ	
ϕ	
$\delta\psi_m$	
$\delta\theta_m$	
p	
q	
r	

10. PLOTTERS

11 x 17 PLOTTERS

PARAMETERS

(1) σ_B vs R_B For Powered Abort Phase only.

(2) ΔV vs θ

30 x 30 PLOTTERS

(1) σ_B vs R_B Polar Plot

(2) R_{xA} vs R_{yA}

R_{xA} vs R_{zA}

PART II

DETAILED DESCRIPTION OF THE SIMULATION AND PROGRAM

1. PHYSICAL DESCRIPTION

The Manned Aerospace Flight Simulator Facility contains a moving base cockpit enclosed in a gondola. The cockpit seat is adjustable in height. A shoulder harness and seat belt are attached to the seat. The gondola is mounted on a three axis gimbal system which provides rotational motion in pitch, yaw and roll. The range of motion available is $\pm 10^{\circ}$, $\pm 20^{\circ}$, and $\pm 10^{\circ}$, respectively. The entire gondola is mounted on a tubular yoke which is pivoted about the pitch axis. The maximum motion is $\pm 90^{\circ}$ at a 40°/sec rate. Two g's is the maximum acceleration obtainable with the simulator. The hydraulic system which drives the moving base is capable of generating up to 4,000 psi. A control console located in the simulator room provides a closed circuit TV visual view of the pilot's head and shoulders. In the event of pilot discomfort, due to excessive motion or computer malfunction, the run can be halted at this console by the safety engineer. Sound effects are simulated by a noise generating system as a function of throttle position for the descent engine. It will be constant for the ascent engine. Located in the computer room is a control engineer's console which has repeater instruments to display information to the experimenter which is displayed to the pilot. An intercom system provides for voice communication between the pilot, computer room and the safety monitor's console.

The gondola is enclosed by a 20 foot hemispherical projection screen. A starfield projector is used to display a starfield on the projection screen. The line-of-sight rates $\dot{A}_B$ and $\dot{E}_B$ are integrated to drive the starfield projector. There is a reset button in the cockpit which centers the starfield if it should pass out of the pilot's line-of-sight. This will occur after 45° of travel. A spot projector will give the CSM position. The projector is driven by the line-of-sight angles with respect to the LEM body axis.= The beam of light from the spot projector will be mechanically interrupted in order to display the CSM position as a flashing light. A horizon and lunar scene are displayed by a projector which is mounted on a four gimbal system. The four gimbal system allows for continuous rotation in roll.

The cockpit is enclosed in a gondola. Directly in front of the pilot is the instrument panel which contains TV and meter displays of parameters to be evaluated in flying an abort mission. Mounted on the left side of the panel are the thruster quad shut-off switches, the Attitude Mode Select Switch, and the Deadband Select Switches. When abort occurs, a red light on the right side of the panel will be illuminated by a computer signal. The throttle-controller is located directly under the lower left side of the instrument panel. The throttle-controller is used to control the thrust level of the descent engine and firing of the reaction control system jets for translational control.

2. INSTRUMENT PANEL

An instrument panel located in the cockpit, provides the pilot with various visual displays. Displayed parameters, their definition and method of display are:

a. Euler Angles:

ϕ, θ, ψ - W/R landing site inertial.

ϕt, θt, ψt - W/R thrust axis.

ϕA, θA, ψA - W/R CSM local vertical.

These angles will drive the attitude ball. Prior to staging θt, ψt, and ϕt will drive the ball. The reference system which is determined prior to a run will determine which of the other two Euler Angle sets shall drive the ball. The angles as displayed to the pilot will be pilot oriented yaw, pitch and roll. These angles will also be digitally displayed as precision angle readouts on TV in the cockpit. The cycle time to display one complete set of angles to the pilot will be 0.5 sec.

b. Attitude Errors:

$$\mathcal{E}_\phi = \phi c - \phi$$
$$\mathcal{E}_\theta = \theta c - \theta$$
$$\mathcal{E}_\psi = \psi c - \psi$$

The subscript c indicates a command function. θ, ϕ and ψ are the instantaneous values of the Euler angles. These error signals are implemented to drive the Flight Director Error Needles located on the Attitude Ball. The experimental matrix will determine which set is utilized. The error needles are scaled to $\pm 0.5^\circ$.

c. Angular Body Rates (p, q, r):

The LEM angular body velocities in roll, pitch and yaw will drive galvanometer type meters. The maximum range is $\pm 20^\circ$/sec with graduations every 5°/sec. The parameters displayed will be pilot oriented such that the yaw rate is equal to "-p" and the roll rate is equal to "r".

d. Altitude Rate:

The altitude rate, $-V_{r1B}$, is an inertial velocity measured with respect to the LEM local vertical. It is displayed on a dial meter with a range of $\pm$ 500 ft/sec.

e. Range Rate:

The range rate $\dot{\rho}$ is the rate of change of the line-of-sight distance between the LEM and the CSM. It is scaled for a maximum of $\pm$ 500 ft/sec. A dial type meter is used for display.

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f. Altitude:

The altitude, $R_B = R_p - R_M$ is displayed on a dial meter from 0 to 100,000 feet full scale. This is the distance of the LEM from the lunar surface as given by the radar altimeter.

g. Thrust to weight ratio:

An acceleration along the X_B LEM body axis. It is displayed on a dial type meter and has a range of 0 to 2 g's (earth g).

h. Elapsed Time:

Displayed on a CRT and reads the total run time in seconds.

i. Set Time:

A CRT display in seconds. The Set Time may be mechanically set to zero by the pilot or stopped during a run. If not reset to zero after being stopped, it will continue to add to the previous value.

j. Elevation Angle, E:

A radar line-of-sight angle measured along the line-of-sight with respect to the LEM body axis. It is defined as the angle the projection of the LOS into the $X_B - Z_B$ plane makes with the positive Z_B axis. It is digitally displayed on TV for a range of $\pm 360^\circ$. The angle E_T is displayed prior to staging.

k. Azimuth Angle, A:

A line-of-sight radar angle which is defined as the angle which the line-of-sight makes with the X_B-Z_B plane. It is digitally read out on TV with a range of $\pm 90^\circ$.

l. Range, ρ :

The distance measured in feet between the LEM and the CSM. It is displayed digitally on TV for a range of 0 to 1,500,000 feet.

m. ΔV USED:

The total amount of ΔV consumed in the ascent and descent engines. It is digitally displayed for a range of 0 to 20,000 feet/sec.

n. ΔV TO GO:

A programmed value of ΔV which is displayed during the thrusting period starting with abort and ending with burnout. It is a digital display on TV. The range is 0 to 9999.9 feet/sec with capability of passing thru zero. The final portion of ΔV TO GO from 100 feet/sec drives a tape called the ΔV TO GO tape meter. This tape has green and gray colors. The gray color will be exposed when ΔV TO GO is in excess of 100 feet/sec and will start moving when

n. ΔV TO GO (Cont'd):

100 feet/sec is reached. The green color will then become visible. At zero ΔV TO GO the line dividing the green and gray color will be aligned with a fixed indicator.

3. COCKPIT SWITCHES

There is an assortment of switches located on the instrument panel. Their function and description are:

a. The Ascent Engine Control Switch:

A push-push type for firing and stopping the ascent engine. This switch is disabled prior to staging. Prior to initial firing of the ascent engine, the stop light is on.

b. Descent Engine Control Switch:

Fires and stops the descent engine. To initiate a run with the descent engine firing, it must be put in the fire position prior to the start of the run. When a programmed failure of the descent engine occurs, the fire position is disabled. Prior to staging it may be used for firing and stopping the descent engine. The switch lights are disabled after staging.

c. "Stage" switch:

Used to allow for staging the descent engine independently of the abort auto sequence switch. When staging is initiated, the Stage light goes on and remains on for the remainder of the run. The Stage switch is disabled while the descent engine is firing.

d. Abort Auto Sequence Switch:

Disabled while the descent engine is in use. It cannot be used after staging of the descent engine nor can it be used more than once in a given run. When used, the switch automatically stages the descent engine and fires the ascent engine. The AAS switch light stays on until the end of the run once it is used.

The Attitude Display switches are all push-push type switches and have back up lights. When one is used, it disables the other attitude display switches. There are five such switches, two of which are non-functional.

e. Descent Display Switch:

Enabled prior to the start of a run except for those runs starting at the insertion thru mid-course phase. It provides for a display of attitude with respect to landing site inertial.

f. Ascent Display Switch:

Must be enabled subsequent to abort and immediately after the attitude mode selector switch is out of the "Auto" mode. Attitude errors will then drive the error needles on the attitude ball if this is desired. Simultaneously, it selects the attitude reference system with respect to landing site inertial or CSM local vertical as determined by the experimental matrix. For this simulation, CSM local vertical is the only system to be displayed for insertion through mid-course.

g. Mid-Course Display Switch:

To be used for the insertion through mid-course portion of a run. It selects the CSM local vertical reference system for altitude display.

h. Hover-Land Switch:

Non-functional for this simulation.

i. Rendezvous Switch:

Non-functional for this simulation.

j. ΔV Set Mode Switch:

This is a two position switch, "Auto" and "Manual". The switch remains in Auto until completion of circularization after which it is placed in manual for insertion into a transfer orbit. While in manual, the pilot may insert a ΔV TO GO by turning the ΔV set wheel.

k. Mid-Course Measure Freeze:

The use of this switch allows readings to be taken on the LOS Angles, range, and range rate for mid-course measurements. Pushing the switch freezes the readout of these functions. The switch must be enabled again for the readout of these parameters to continue.

l. Thruster-Quad Shut-Off:

Allows for manual shut-off of any one of four RCS thruster quads by the pilot. When energized, the appropriate fail light goes on.

m. Attitude:

There are three attitude switches which enable the pilot to select either direct on-off or "normal" for pitch, roll and yaw axes independently.

n. Mode Select Switch:

This is a three position switch, "Auto," "Attitude Hold," and "Rate Command." The switch is placed in the "Auto" mode prior to a run and remains there until Abort is initiated.

o. Dead Band Select Switch:

This switch enables the pilot to select either maximum or minimum dead-band for the control system. The selection is made for all three axes simultaneously.

p. Translation Select:

This switch is used to select either direct on-off or Pulse Train Modulation for the translational mode.

4. INERTIAL PITCH PROGRAM

An inertial pitch, and a Line-Of-Sight (LOS) pitch program will be studied in this simulation. Studies will be conducted using ascent engine only and an ascent/descent engine combination for each program.

The inertial pitch program with ascent engine only will require the pilot to rotate the vehicle, at abort, to an inertial pitch angle of 0^o at a 10^o/sec rate. The inertial pitch angle is referenced to landing site local horizontal. The yaw and roll angles should be simultaneously held at zero. The ascent engine is then fired and the vehicle held at the inertial pitch angle of 0^o until a specified value of ΔV USED is attained. The vehicle is then pitched to a specified pitch angle, θ, at a 10^o/sec rate. This angle is held constant until a new specified value of ΔV USED is attained. The vehicle is then pitched at 10^o/sec to a second specified pitch angle, θ_2. This procedure is repeated until ΔV USED burnout is attained. The ascent engine is then stopped and the vehicle coasts in a circular parking orbit of altitude h_{BO}. After coasting in this orbit for a period of time, the ascent engine is fired and the vehicle inserted into a coasting transfer orbit by thrusting at an inertial pitch angle θ_T, until ΔV USED has been attained. The period of time spent in the circular parking orbit will depend upon the time at which abort occurs since this will affect the phasing of the LEM and CSM. This time may range from 2 to 20 minutes. During the mid-course phase the LEM will nominally be on a collision course with the CSM and no mid-course corrections would be required. Due to pilot induced errors mid-course corrections will be made as required.

The procedure is the same when the ascent/descent engines are used except that the descent engine remains on at abort. The descent engine thrust at this time is 10,500 pounds. When ΔV USED equals 5,000 ft/sec, the descent engine is throttled back to 6,000 pounds. This thrust level is maintained until descent engine burnout or staging of the descent engine occurs. The thrust of 6,000 pounds was chosen to make the thrust to weight ration, T/W, equivalent to the T/W of the ascent engine at abort. The descent engine will be staged when ΔV USED equals 7,400 ft/sec. The ascent engine is then fired. Tables of the inertial pitch program for both engine configurations are included in the Appendix. The procedures, the same for both engine configurations, are included in the Appendix. The procedures are the same for both nominal and off-nominal trajectories.

5. LOS PITCH PROGRAM

The procedure to be followed in the LOS pitch program is similar to the inertial pitch program. At abort, the vehicle is rotated until the LOS elevation angle E_1 is attained. For late abort cases the thrust will be applied along the local vertical direction. In some cases, a rotation of 180^o in ϕ will be necessary in order to satisfy radar look angle requirements. The yaw and roll angles will be held to zero during the pitch rotation to E_1. The angle E_1 is held constant until a specific value of ΔV USED is attained. The vehicle is then pitched to an LOS angle E_2 and held at this angle until ΔV USED burnout is attained. The vehicle will then be in a circular parking orbit. After coasting for a specified period of time, the ascent engine is ignited and the vehicle placed in a coasting transfer orbit by thrusting and maintaining an LOS angle E_T until ΔV USED $_T$ has been attained. The procedure is the same from this point as in the inertial pitch program.

6. EXPERIMENTAL TEST PLAN

a. General

In this section are contained the basic information and requirements of those LEM engineering groups that have requested data from the Manual Abort Simulation program. The number of problems posed by these groups precludes their resolution within the six-week experimental period. Thus, a priority has been established to achieve maximum possible benefit from this simulation study.

b. Purpose

The major objective of this investigation is to determine pilot ability to perform manual abort within the present LEM ΔV budget. Additional objectives of the study, in order of priority, are:

(1) Determine required attitude instrumentation for manual abort.

(2) Determine effectiveness of manual midcourse corrections.

(3) Determine the effect of reaction control system degradation on manual abort capability and ΔV.

c. Subject Program

Each pilot will be subjected to the following program:

(1) Indoctrination.

(2) Training Trials.

(3) Pre-test Trials.

(4) Experimental Test Trials.

d. Pilot Indoctrination

In the indoctrination phase, subjects will be given a verbal description of the Abort problem, the limitations of the simulation, the flight control system and its dynamics, the vehicle configurations to be used, and the criteria to be met for each part-task and full-task segment. Written instructions concerning the tasks required will also be given. Subjects will be familiarized with the physical components of the simulator and its handling characteristics. Critical points in the trajectory will be shown to the subjects as well as how the displays function during the dynamic situation.

e. Pilot Training Trials

(1) Abort-to-Burnout Training Trails.

The conditions to be employed in these training runs (abort-to-transfer orbit insertion) will be as follows:

(a) Inertial pitch program for all trials.

(b) The ΔV USED and ΔV TO GO digital indicators and associated ΔV TO GO vernier display will be operational for all trials.

(c) Control deadband of 0.1° will be used.

(d) Abort point #2 along a nominal descent trajectory will be utilized for these runs. Each run will start either 5, 10 or 15 seconds prior to abort time, randomly assigned.

(e) Error needle commands in all 3 axes with respect to inertial reference system during abort ascent trajectory and coasting parking orbit.

Each of 4 pilots will undergo training trials using the ascent engine only and descent/ascent engine combination until his performance meets the specified criterion of 2 consecutive successful runs per engine configuration. Training qualification will be based upon ΔV and terminal conditions at burnout.

Upon completion of the abort-to-burnout training phase, subjects will then undergo a series of post-training trials, as presented in Table 1.

(2) Mid-Course Correction Training Trials.

These runs will follow the successful completion of abort-to-burnout training. The following conditions will be held constant throughout the midcourse correction (i.e., transfer orbit insertion through midcourse thrusting) training trials:

(2) Mid-Course Correction Training Trials (Cont'd).

(a) CSM local vertical reference system for all trials.

(b) The ΔV USED and ΔV TO GO digital indicators and associated ΔV TO GO vernier display will be operational for all trials.

(c) Control deadband of 0.1° will be used.

(d) The initial conditions for all midcourse correction training trials are presented in Table 2. Each mid-course run will begin at transfer orbit insertion.

(e) Thrusting procedures will be accomplished with dual-axis control, i.e., rotation about the Y-body axis followed by successive thrusting along the X and Y axes.

(f) Error needle commands in roll, pitch and yaw will be utilized for mid-course measurements and both the precision angle digital readouts in all three axes and the eight-ball will be used for mid-course thrusting.

Training criteria for mid-course correction procedures will be satisfied after each pilot performs two consecutive successful runs, based upon ΔV and burnout errors.

f. Pre-Test Trials

This phase of the simulation program will be used to evaluate attitude display instrumentation, reaction control system sensitivity (deadbands) and the attitude reference system (landing site inertial vs line-of-sight reference systems). Based upon these evaluations, certain quantities will be selected and remain constant for the remainder of the program.

(1) Attitude Display Instrumentation (Abort-to-Transfer Orbit Insertion)

This segment of the pre-test trials will investigate two attitude instrumentation schemes (i.e., precision angle readouts and eight-ball only) to determine the feasibility of their inclusion as study variables in the experimental test runs. Should both attitude display modes prove "flyable," they will be retained for the balance of the study.

The following conditions will be held constant throughout these runs:

(a) Inertial pitch program will be used.

(1) Attitude Display Instrumentation (Cont'd).

(b) The ΔV USED and ΔV TO GO digital indicators and associated ΔV TO GO vernier display will be operational.

(c) Control deadband of 0.1° will be utilized.

(d) Abort point #2 along a nominal descent trajectory will be employed for these runs. Each run will start either 5, 10 or 15 seconds prior to abort time.

Display conditions involving precision angle digital readouts in all three axes with the eight-ball and the eight-ball only (no precision angle readouts) will be introduced. Each of four conditions will be presented once to two pilots as presented in Table 3.

Should one display condition, or both, prove to be "flyable" (based upon ΔV and burnout errors), they will be retained and treated as study variables during the test runs. If there is no initial success, the run schedule (Table 3) will be repeated three additional times per pilot, i.e., a total of 16 runs per pilot. At least eight "non-flyable" runs (i.e., in excess of ΔV budget) per pilot under each display method would be used in justifying the elimination of one, or both.

(2) Deadband Investigation.

Three different control deadband conditions will be studied to determine the effect of reaction control system sensitivity on pilot performance during abort-to-transfer orbit insertion, based upon ΔV and burnout errors. The study conditions to be employed for these runs will be identical to those used in the abort-to-burnout training trials (refer to section e(1)) with the exception of control deadband conditions. Each of three values of control deadband will be introduced, i.e., 0.05°, 0.5° and 1.0° under each of two engine configurations. Table 4 presents the six conditions that will be administered to each of two pilots.

(3) Reference System Evaluation (Abort-to-transfer orbit insertion).

Two attitude reference coordinate systems will be investigated to determine the superior technique for inclusion in the test runs. The abort-to-burnout trajectory will be referenced to either the landing site inertial or the line-of-sight coordinate system. These pre-test trials will be conducted using both reference systems and the ΔV consumed for each will be recorded. The ΔV USED for the two referenced systems will be compared to determine if any significant difference between the two referenced techniques exists. If one is significant, the superior referenced technique, i.e., the system requiring the least ΔV, will be retained for the remainder of the program.

(3) Reference System Evaluation (Cont'd).

The study conditions to be employed for these runs will be identical to those used in the abort-to-burnout training trials (refer to section e(1)) with the exception of the reference technique used, i.e., line-of-sight (LOS) reference system instead of landing sight inertial. Error needle commands will be employed during the abort ascent trajectory and coasting parking orbit. The angles to be displayed on the error needles are: Inertial Roll Angle, Inertial Yaw Angle and Elevation Angle. Each of two pilots will fly ascent engine only and descent/ascent engine combination until they complete two consecutive "successful" runs per engine configuration (based upon the same criteria of ΔV and terminal conditions at burnout used in section e(1). The pilots will perform two runs with the ascent engine only and two runs with the descent/ascent engine combination as presented in Table 5. The ΔV and terminal conditions at burnout will be recorded and compared with pilot performance data using the inertial reference system (refer to section e(1)) to determine if any significant difference between the two referenced techniques exists. If one is significant, it will be retained for use in the test runs. If there is no difference, the inertial technique will be retained.

g. Experimental Test Trials

This phase of the simulation program will be used to evaluate pilot capability in (1) the manual initiation and control of powered abort trajectories under a variety of vehicle and attitude display conditions; (2) the conduct of a manual midcourse correction, and; (3) the manual control of a complete abort trajectory (abort through mid-course correction) under a variety of conditions of flight control system degradation. The test runs will be sub-divided into three segments as follows:

. Abort-to-burnout.

. Mid-course correction.

. Complete abort trajectory.

(1) Abort-to-Burnout Test Trials.

The conditions to be employed in these test runs (abort-to-transfer orbit insertion) will be as follows:

(a) Either inertial or LOS pitch program will be used, as determined in section f(3).

(b) The ΔV USED and ΔV TO GO digital indicators and associated ΔV TO GO vernier display will be operational for all trials.

(c) Control deadband of 0.1° will be used for all trials.

(d) Each test trial will start either 5, 10 or 15 seconds prior to abort time, randomly assigned.

The study variables under investigation are as follows:

(1) Abort-to-Burnout Test Trials (Cont'd).

Attitude Display Instrumentation (Assume all three are found feasible - refer to section f(1)).

E.N. = Error Nulling Needles.

P.A. = Precision angle readouts in three axes with eight-ball (if inertial is used); Precision angle readouts in roll, yaw and LOS elevation with eight-ball (if LOS is used).

E.B. = Eight-ball only.

Abort Points

1N = Abort point #1 along nominal descent trajectory where $t_{ABORT} = 40$ seconds after initiation of powered descent;and $\theta_{ABORT} = 97.9^{o}$.

3N = Abort point #3 along nominal descent trajectory ($t_{ABORT} = 280$ seconds; $\theta_{ABORT} = 75.6^{o}$).

5N = Abort point #5 along nominal descent trajectory ($t_{ABORT} = 510.5$ seconds; $\theta_{ABORT} = 0.1^{o}$).

3 ON = Abort Point #3 along off-nominal descent trajectory.

Engine Configurations

A = Ascent engine only.

D/A = Descent/Ascent engine combination.

Each of the 24 conditions will be presented to each of four pilots twice, yielding a total of 48 runs per subject. Table 6 presents the run schedule for these test trials.

(2) Mid-Course Test Trials.

The test conditions for this series of trials will be as follows:

(a) CSM local vertical reference system will be used for all trials.

(b) The ΔV USED and ΔV TO GO digital indicators and associated ΔV TO GO vernier display will be operational for all trials.

(c) Control deadband of 0.1^{o} will be used.

(d) Each test trial will begin at transfer orbit insertion.

(2) Mid-Course Test Trials (Cont'd).

(e) Mid-course correction thrusting procedures for coplanar errors only will be accomplished by rotating about the Y-body axis followed by a single translational thrust application along the X-body axis. When correcting out-of-plane errors, the pilot will rotate about the Y-body axis followed by successive translational thrusting along the X- and Y-body axes.

The study variables under investigation are as follows:

Attitude Display Instrumentation (Assume all three are feasible)

a = Error needles in all three axes for mid-course measurement and precision angle digital readouts in all three axes with eight-ball for mid-course thrusting.

b = Precision angle digital readouts in all three axes with eight-ball for both mid-course measurements and thrusting.

c = Eight-ball only for both mid-course measurements and thrusting.

Initial Conditions (Will be derived from pilot performance data, i.e., burnout conditions at transfer orbit insertion, collected during abort-to-burnout training trials)

1 = "Moderate" co-planar errors.

2 = "Extreme" co-planar errors.

3 = "Moderate" co-planar and out-of-plane errors.

4 = "Extreme" co-planar and out-of-plane errors.

Four pilots will undergo each of the twelve test conditions twice, yielding a total of 24 runs per subject. The run schedule for the mid-course test trials are presented in Table 7.

(3) Complete Abort Trajectory Test Trials.

The study conditions to be employed for these runs will be as follows:

(a) For the abort-to-transfer orbit insertion portion of each run, either the Inertial or LOS pitch program will be used, whichever is found superior (refer to section f(3)).

(3) Complete Abort Trajectory Test Trials (Cont'd).

(b) The superior attitude display will be utilized, as determined in section g(1), for the abort-to-transfer orbit insertion segment of each test trial.

(c) The ΔV USED and ΔV TO GO digital indicators and associated ΔV TO GO vernier display will be operational for all trials.

(d) Control deadband of 0.1^{o} will be utilized.

(e) Abort point #4 along a nominal descent trajectory will be used for these runs. Each run will start either 5, 10 or 15 seconds prior to abort time, randomly assigned.

(f) CSM local vertical reference system will be used for all trials for the transfer orbit insertion through mid-course thrusting segment.

(g) The attitude display instrumentation found to be superior in section g(2) will be used during the mid-course correction portion of each test run.

A variety of degraded conditions of the flight control and reaction control systems will be introduced at the beginning of each run. The study variables under investigation are as follows:

<u>Vehicle Configurations</u>

A = Ascent engine only.

D/A = Descent/Ascent engine combination.

<u>Flight Control Degradation</u>

N = No degradation or malfunction.

D.P. = Direct mode in pitch axis only.

D = Direct mode in all three axes.

C = Rate command in all 3 axes; no attitude hold.

R.O. = One of the RCS jets (used for pitch, i.e., x-axis jet) failed <u>on</u>.

R.F. = One of the RCS jets (used for pitch) failed <u>off</u>.

Four pilots will undergo each of the twelve test conditions once, yielding a total of twelve runs per subject. The run schedule for the complete abort trajectory test trials are presented in Table 8.

APPENDIX A

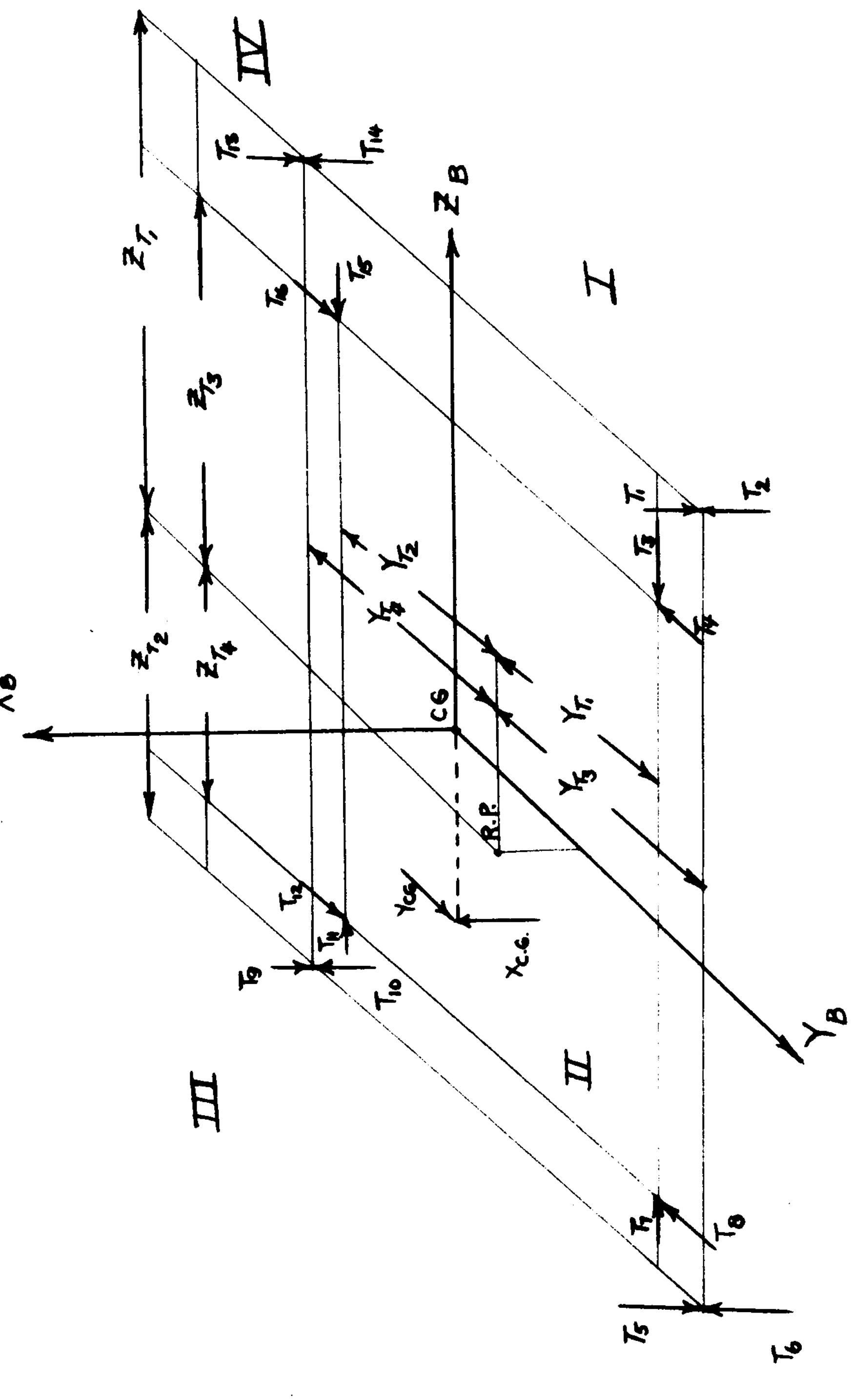
EQUATIONS OF MOTION

The equations of motion developed by the Dynamic Analysis Group, are for a six degrees of freedom hybrid simulation. Except for the LEM body axis angular rates p, q and r, the implementation of the equations is digital. The assumptions made in formulating the equations are as follows:

1. The CSM is in a circular equatorial orbit.
2. The Descent Engine gimbal axis excursions are small.
3. The Ascent Engine thrust misalignment is small and constant.
4. The reaction jets have no thrust misalignment.
5. There is no reaction between the LEM Ascent and Descent stages at separation.
6. There is no movement or sloshing of fuel in the LEM.
7. The moon has no rotation with respect to inertial space.

The translational and rotational equations of motion include the effects of descent engine gimballing,ascent engine thrust misalignment, inertial cross-coupling and jet damping. The mass and inertia equations include the effect of staging after descent engine burnout. The equations provide for a variable center of gravity with moments measured from a fixed reference point. The moments are then translated and are taken about the center of gravity. The reaction jet moment arm geometry is as shown in the following figure: (See page A-2).

LEM REACTION JET GEOMETRY



CSM TRANSLATIONAL MOTION (INERTIAL)

$$\ddot{X}_{PA} = - \frac{G_M}{R_A^3} X_{PA}$$

$$\ddot{Y}_{PA} = 0$$

$$\ddot{Z}_{PA} = - \frac{G_M}{R_A^3} Z_{PA}$$

RELATIVE POSITION WRT INERTIAL AXES

$$R_{xp} = X_{pA} - X_p$$

$$R_{yp} = - Y_p$$

$$R_{zp} = Z_{pA} - Z_p$$

RELATIVE POSITION IN LEM BODY AXES

$$R_{xB} = l_1 R_{xp} + l_2 R_{yp} + l_3 R_{zp}$$

$$R_{yB} = m_1 R_{xp} + m_2 R_{yp} + m_3 R_{zp}$$

$$R_{zB} = n_1 R_{xp} + n_2 R_{yp} + n_3 R_{zp}$$

RANGE OF LEM TO CSM

$$\rho = (R_{xp}^2 + R_{yp}^2 + R_{zp}^2)^{1/2}$$

LEM INERTIAL VELOCITY WRT LEM LOCAL VERTICAL

$$\dot{X}_R = \dot{X}_p \cos \sigma_B + \dot{Z}_p \sin \sigma_B$$

$$\dot{X}_{n1B} = -\dot{X}_p \sin \sigma_B + \dot{Z}_p \cos \sigma_B$$

$$V_{n2B} = \dot{X}_R \sin \lambda_B + \dot{Y}_p \cos \lambda_B$$

$$V_{R1B} = -\dot{X}_R \cos \lambda_B + \dot{Y}_p \sin \lambda_B$$

LEM LATITUDE AND LONGITUDE RATES

$$\dot{\lambda}_B = -\frac{V_{n2B}}{R_p}$$

$$\dot{\sigma}_B = \frac{V_{n1B}}{X_R}$$

LEM ALTITUDE

$$X_R = (X_p^2 + Z_p^2)^{1/2}$$

$$R_p = (X_R^2 + Y_p^2)^{1/2}$$

$$R_B = R_p - R_M$$

LEM LATITUDE AND LONGITUDE

$$\lambda_B = \sin \lambda_B = -Y_p/R_p$$

$$\cos \lambda_B = X_R/R_p$$

$$\sin \sigma_B = Z_p/X_R$$

$$\cos \sigma_B = X_p/X_R$$

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LEM BODY AXES FORCE EQUATIONS

$$B_x = (T_2 + T_6 + T_{10} + T_{14}) - (T_1 + T_5 + T_9 + T_{13}) + T_m + T_a$$

$$B_y = (T_{12} + T_{16}) - (T_4 + T_8) + \delta\psi_m T_m + \delta\psi_a T_a$$

$$B_z = (T_7 + T_{11}) - (T_{13} + T_{15}) + \delta\theta_m T_m + \delta\theta_a T_a$$

LEM RESULTANT INERTIAL ACCELERATION FORCES

$$\Delta\ddot{X}_p = \frac{(l_1 B_x + m_1 B_y + n_1 B_z)}{m_T}$$

$$\Delta\ddot{Y}_p = \frac{l_2 B_x + m_2 B_y + n_2 B_z}{m_T}$$

$$\Delta\ddot{Z}_p = \frac{B_x l_3 + m_3 B_y + n_3 B_z}{m_T}$$

INERTIAL TRANSLATIONAL MOTION

$$\ddot{X}_p = \Delta\ddot{X}_p - \frac{G_m}{R_p^3} X_p$$

$$\ddot{Y}_p = \Delta\ddot{Y}_p - \frac{G_m}{R_p^3} Y_p$$

$$\ddot{Z}_p = \Delta\ddot{Z} - \frac{G_m}{R_p^3} Z_p$$

LEM BODY AXES MOMENT EQUATIONS

$$L = \left\{(T_4 + T_7 + T_{12} + T_{15}) - (T_3 + T_8 + T_{11} + T_{16})\right\} Y_T + B_y Z_{CG} - B_Z Y_{CG}$$

$$M = \left\{(T_2 + T_5 + T_9 + T_{14}) - (T_1 + T_6 + T_{10} + T_{13})\right\} Y_{T_3} - B_X Z_{CG} + B_Z Y_{CG}$$
$$+ \delta_{\theta m} X_{Tm} T_m + \delta_{\theta a} X_{Ta} T_a$$

$$N = \left\{(T_1 + T_5 + T_{10} + T_{14}) - (T_2 + T_6 + T_9 + T_{13})\right\} Y_{T_3} + Y_{CG} B_X - X_{CG} B_Y$$
$$- \delta_{\psi m} X_{Tm} T_m - \delta_{\psi a} X_{Ta} T_a$$

ROTATIONAL MOTION

$$\dot{p} = \frac{1}{I_x} \left\{ L - (I_z - I_y)\, qr + I_{yz} (q^2 - r^2) + I_{yz} (\dot{r} + pq) \right.$$
$$\left. + I_{xy} (\dot{q} - pr) + p\dot{M}_m\, l^2_{Dp_D} + p\dot{M}_a\, l^2_{Dp_A} \right\}$$

$$\dot{q} = I_y \left\{ M - (I_x - I_z)\, pr + I_{xz} (r^2 - p^2) + I_{xy} (\dot{p} + qr) + I_{yz} (\dot{r} - pq) \right.$$
$$\left. + q\dot{M}_m\, l^2_{Dqr_D} + q\dot{M}_a\, l^2_{Dqr_A} \right\}$$

$$\dot{r} = \frac{1}{I_z} \left\{ N - (I_y - I_x)\, pq + I_{xy} (p^2 - q^2) + I_{yz} (\dot{q} + pr) + I_{xz} (\dot{p} - qr) \right.$$
$$\left. + r\dot{M}_m\, l^2_{Dqr_D} + r\dot{M}_a\, l^2_{Dqr_A} \right\}$$

LINE OF SIGHT ANGLES WRT LEM BODY AXES

$$\text{Sin } E = R_{XB}/\rho$$

$$\text{Cos } E = (R_{YB}{}^2 + R_{ZB}{}^2)^{1/2}$$

$$\text{Sin } A = - R_{YB}/\rho \text{ Cos } E$$

$$\text{Cos } A = R_{ZB}/\rho \text{ Cos } E$$

RELATIVE VELOCITY OF CSM WRT INERTIAL AXES

$$U_{xp} = \dot{X}_{pA} - \dot{X}_p$$

$$U_{yp} = - \dot{Y}_p$$

$$U_{zp} = \dot{Z}_{pA} - \dot{Z}_p$$

RELATIVE VELOCITY OF CSM WRT LEM BODY AXES

$$U_{xB} = l_1 U_{xp} + l_2 U_{yp} + l_3 U_{zp}$$

$$U_{yB} = m_1 U_{xp} + m_2 U_{yp} + m_3 U_{zp}$$

$$U_{zB} = n_1 U_{xp} + n_2 U_{yp} + n_3 U_{zp}$$

LOS RATES WRT LEM BODY AXES

$$\dot{\rho} = U_{xB} \operatorname{Sin} E - U_{yB} \operatorname{Cos} E \operatorname{Sin} A + U_{zB} \operatorname{Cos} E \operatorname{Cos} A$$

$$\dot{E} = \frac{1}{\rho}(U_{xB} \operatorname{Cos} E + U_{yB} \operatorname{Sin} E \operatorname{Sin} A - U_{zB} \operatorname{Sin} E \operatorname{Cos} A)$$

$$\dot{A} = \frac{1}{\rho \operatorname{Cos} E} (-U_{yB} \operatorname{Cos} A - U_{zB} \operatorname{Sin} A)$$

RELATIVE POSITION OF LEM WRT CSM LOCAL VERTICAL

$$R_{xA} = R_{xp} \operatorname{Sin} \sigma_A - R_{zp} \operatorname{Cos} \sigma_A$$
$$R_{yA} = -R_{yp}$$
$$R_{zA} = R_{xp} \operatorname{Cos} \sigma_A + R_{zp} \operatorname{Sin} \sigma_A$$

LOS ANGLES WRT CSM LOCAL VERTICAL AXES

$$\operatorname{Sin} E_A = - R_{zA}/\rho \operatorname{Cos} A \qquad \operatorname{Sin} A_A = R_{yA}/\rho$$

$$\operatorname{Cos} E_A = R_{xA}/\rho \operatorname{Cos} A \qquad \operatorname{Cos} A_A = (R_{xA}{}^2 + R_{zA}{}^2)^{1/2}$$

FUEL CONSUMPTION EQUATIONS

$$\dot{M}_m = T_m/32.2\ I_{spm}$$

$$\dot{M}_a = T_a/32.2\ I_{spa}$$

$$\Delta M_m = \int_o^t \dot{M}_m dt$$

$$\Delta M_a = \int_o^t \dot{M}_a dt$$

$$\Delta M_A = \frac{1}{32.2\ I_{spj}} \int_o^t (T_1+T_3+T_6+T_7+T_9+T_{12}+T_{14}+T_{16}) dt$$

$$\Delta M_B = \frac{1}{32.2\ I_{spj}} \int_o^t (T_2+T_4+T_5+T_8+T_{10}+T_{11}+T_{13}+T_{15}) dt$$

$$\Delta M_{RCS} = \Delta M_a + \Delta M_B$$

$$\Delta M_T = \Delta M_T$$

$$\Delta M_{T_1} = \Delta M_n + \Delta M_{RCS}$$

$$\Delta M_{T_2} = \Delta M_a + \Delta M_{RCS}$$

$$\Delta M_{jx+} = \frac{1}{32.2\ I_{spj}} \int_o^t (T_2 + T_6 + T_{10} + T_{14}) dt$$

$$\Delta M_{jx-} = \frac{1}{32.2\ I_{spj}} \int_o^t (T_1 + T_5 + T_9 + T_{13}) dt$$

$$\Delta M_{jy+} = \frac{1}{32.2\ I_{spj}} \int_o^t (T_{12} + T_{16}) dt$$

FUEL CONSUMPTION EQUATIONS (Cont'd)

$$\Delta M_{jy-} = \frac{1}{32.2\ I_{spj}} \int_0^t (T_4 + T_8)dt$$

$$\Delta M_{jz+} = \frac{1}{32.2\ I_{spj}} \int_0^t (T_7 + T_{11})dt$$

$$\Delta M_{jz-} = \frac{1}{32.2\ I_{spj}} \int_0^t (T_3 + T_{15})dt$$

$$\Delta V_{TR} = \int_0^t \frac{1}{M_T} \left(|B_x| + |B_y| + |B_z| \right)dt$$

$$\Delta V_m = \int_0^t T_m/M_T\ dt$$

$$\Delta V_a = \int_0^t \frac{T_a}{M_T}\ dt$$

INSTANTANEOUS MASS

$$M_T = M_{T_1} \text{ or } M_{T_2}$$

$$M_{T_1} = M_{oT_1} - \Delta M_{T_1}$$

$$M_{T_2} = M_{oT_2} - \Delta M_{T_2}$$

DIRECTION COSINES

LEM BODY TO INERTIAL

$$\begin{bmatrix} \dot{l}_1 & \dot{m}_1 & \dot{n}_1 \\ \dot{l}_2 & \dot{m}_2 & \dot{n}_2 \\ \dot{l}_3 & \dot{m}_3 & \dot{n}_3 \end{bmatrix} = \begin{bmatrix} l_1 & m_1 & n_1 \\ l_2 & m_2 & n_2 \\ l_3 & m_3 & n_3 \end{bmatrix} \cdot \begin{bmatrix} 0 & -r & q \\ r & 0 & -p \\ -q & p & 0 \end{bmatrix}$$

LEM BODY TO CSM LOCAL VERTICAL

$$\begin{bmatrix} -L_3 & -M_3 & -N_3 \\ L_2 & M_2 & N_2 \\ L_1 & M_1 & N_1 \end{bmatrix} = \begin{bmatrix} \cos \sigma_A & 0 & \sin \sigma_A' \\ 0 & 1 & 0 \\ -\sin \sigma_A' & 0 & \cos \sigma_A \end{bmatrix} \cdot \begin{bmatrix} l_1 & m_1 & n_1 \\ l_2 & m_2 & n_2 \\ l_3 & m_3 & n_3 \end{bmatrix}$$

LEM BODY TO LANDING SITE VERTICAL AXES

$$\begin{bmatrix} l_1' & m_1' & n_1' \\ l_2' & m_2' & n_2' \\ l_3' & m_3' & n_3' \end{bmatrix} = \begin{bmatrix} \cos \sigma_L & 0 & \sin \sigma_L \\ 0 & 1 & 0 \\ -\sin \sigma_L & 0 & \cos \sigma_L \end{bmatrix} \cdot \begin{bmatrix} l_1 & m_1 & n_1 \\ l_2 & m_2 & n_2 \\ l_3 & m_3 & n_3 \end{bmatrix}$$

LEM BODY TO LEM LOCAL VERTICAL

$$\begin{bmatrix} -L_3 & -M_3' & -N_3 \\ L_2' & M_2' & N_2' \\ L_3' & M_3' & N_3' \end{bmatrix} = \begin{bmatrix} \cos \lambda_B & -\sin \lambda_B & 0 \\ \sin \lambda_B & \cos \lambda_B & 0 \\ 0 & 0 & 1 \end{bmatrix} \cdot \begin{bmatrix} l_1 & m_1 & n_1 \\ l_2 & m_2 & n_2 \\ l_3 & m_3 & n_3 \end{bmatrix}$$

DESCENT ENGINE THRUST TO LANDING SITE VERTICAL

$$\begin{bmatrix} l_{1T} & m_{1T} & n_{1T} \\ l_{2T} & m_{2T} & n_{2T} \\ l_{3T} & m_{3T} & n_{3T} \end{bmatrix} = \begin{bmatrix} l_1' & m_1' & n_1' \\ l_2' & m_2' & n_2' \\ l_3' & m_3' & n_3' \end{bmatrix} \cdot \begin{bmatrix} 1 & 0 & -\delta_{\theta m} \\ 0 & 1 & 0 \\ \delta_{\theta m} & 0 & 1 \end{bmatrix} \cdot$$

DESCENT ENGINE THRUST TO LANDING SITE VERTICAL (cont'd)

$$\cdot \begin{bmatrix} 1 & -\delta\psi_m & 0 \\ \delta\psi_m & 1 & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

INERTIA EQUATIONS

$$I_{x1} = I_{x_{01}} - \frac{(dI_x)}{(dM_T)_1} \Delta M_{T1}$$

$$I_{x2} = I_{x_{02}} - \frac{(dI_x)}{(dM_T)_2} \Delta M_{T2}$$

$$I_{y1} = I_{y_{01}} - \frac{(dI_y)}{(dM_T)_1} \Delta M_{T1}$$

$$I_{y2} = I_{y_{02}} - \frac{(dI_y)}{(dM_T)_2} \Delta M_{T2}$$

$$I_{z1} = I_{z_{01}} - \frac{(dI_z)}{(dM_T)_1} \Delta M_{T1}$$

$$I_{z2} = I_{z_{02}} - \frac{(dI_z)}{(dM_T)_2} \Delta M_{T2}$$

$$I_{xy_1} = I_{xy_{01}} - \frac{(dI_{xy})}{(dM_T)_1} \Delta M_{T1}$$

$$I_{xy_2} = I_{xy_{02}} - \frac{(dI_{xy})}{(dM_T)_2} \Delta M_{T2}$$

$$I_{xz_1} = I_{xz_{01}} - \frac{(dI_{xz})}{(dM_T)_1} \Delta M_{T1}$$

$$I_{xz_2} = I_{xz_{02}} - \frac{(dI_{xz})}{(dM_T)_2} \Delta M_{T2}$$

$$I_{yz_1} = I_{yz_{01}} - \frac{(dI_{yz})}{(dM_T)_1} \Delta M_{T1}$$

$$I_{yz_2} = I_{yz_{02}} - \frac{(dI_{yz})}{(dM_T)_2} \Delta M_{T2}$$

CENTER OF GRAVITY

$$X_{CG_1} = X_{CG_{01}} - \left(\frac{dX_{CG}}{dM_T}\right)_1 \Delta M_T$$

$$X_{CG_2} = X_{CG_{02}} - \left(\frac{dX_{CG}}{dM_T}\right)_2 \Delta M_{T_2}$$

$$Y_{CG_1} = Y_{CG_{01}} - \left(\frac{dY_{CG}}{dM_T}\right)_1 \Delta M_{T_1}$$

$$Y_{CG_2} = Y_{CG_{02}} \left(\frac{dY_{CG}}{dM_T}\right)_2 \Delta M_{T_2}$$

$$Z_{CG_1} = Z_{CG_{01}} \left(\frac{dZ_{CG}}{dM_T}\right)_1 \Delta M_{T_1}$$

$$Z_{CG_2} = Z_{CG_{02}} \left(\frac{dZ_{CG}}{dM_T}\right)_2 \Delta M_{T_2}$$

EULER ANGLES

BODY AXES WRT LANDING SITE VERTICAL

$$\text{Sin } \psi = l_2'$$

$$\text{Cos } \psi = + \left[1 - (l_2')^2 \right]^{1/2}$$

$$\text{Sin } \theta = \frac{-\, l_3'}{\text{Cos } \psi} \qquad \text{Cos } \theta = \frac{l_1'}{\text{Cos } \psi}$$

$$\text{Sin } \phi = \frac{-\, n_2'}{\text{Cos } \psi} \qquad \text{Cos } \phi = \frac{m_2'}{\text{Cos } \psi}$$

THRUST AXES ANGLES WRT LANDING SITE VERTICAL

$$\text{Sin } \psi_T = l_{2T} \qquad \text{Sin } \theta_T = -l_{3T} / \text{Cos } \psi_T$$

$$\text{Cos } \psi_T = (1 - l_{2T}^{\,2})^{1/2} \qquad \text{Cos } \theta_T = l_{1T} / \text{Cos } \psi_T$$

$$\text{Sin } \phi_T = -n_{2T} / \text{Cos } \psi_T \qquad \text{Cos } \phi_T = m_{2T} / \text{Cos } \psi_T$$

BODY AXES WRT CSM LOCAL VERTICAL

$$\text{Sin}\ \psi_A = L_2$$

$$\text{Cos}\ \psi_A = +\ (1 - L_2^{\,2})^{1/2}$$

$$\text{Sin}\ \phi_A = -N_2/\text{Cos}\ \psi_A$$

$$\text{Cos}\ \phi_A = M_2/\text{Cos}\ \psi_A$$

$$\text{Sin}\ \theta_A = -L_1/\text{Cos}\ \psi_A$$

$$\text{Cos}\ \theta_A = -L_3/\text{Cos}\ \psi_A$$

EQUATIONS FOR ATTITUDE BALL DRIVE

$$\text{Sin}\ \psi_{BALL} = -\ \text{Sin}\ \phi\ \text{Cos}\ \psi$$

$$\text{Cos}\ \psi_{BALL} = +\ (1 - \text{Sin}^2\ \psi_{BALL})^{1/2}$$

$$\text{Sin}\ \theta_{BALL} = \frac{\text{Cos}\ \phi\ \text{Sin}\ \theta + \text{Sin}\ \phi\ \text{Sin}\ \psi\ \text{Cos}\ \theta}{\text{Cos}\ \psi_{BALL}}$$

$$\text{Cos}\ \theta_{BALL} = \frac{\text{Cos}\ \phi\ \text{Cos}\ \theta - \text{Sin}\ \phi\ \text{Sin}\ \psi\ \text{Sin}\ \theta}{\text{Cos}\ \psi_{BALL}}$$

$$\text{Sin}\ \phi_{BALL} = \frac{\text{Sin}\ \psi}{\text{Cos}\ \psi_{BALL}}$$

$$\text{Cos}\ \phi_{BALL} = \frac{\text{Cos}\ \phi\ \text{Cos}\ \psi}{\text{Cos}\ \psi_{BALL}}$$

ΔV TO GO

$$\Delta V\ \text{TO GO} = K_1 \left\{ \Delta V_{SET} - \int_o^t \frac{1}{M_T} \left(|B_x| + |B_y| + |B_z| \right) dt \right\} + K_2 \left\{ K_3\ \Delta V_{Boa} + K_4\ \Delta V_{Bod} + K_5\ (7400) + K_6\ (7400 - \Delta V_m) - \Delta V_{USED} \right\}$$

ΔV_{SET} is a manual set value

t_{SET} is time of manual set

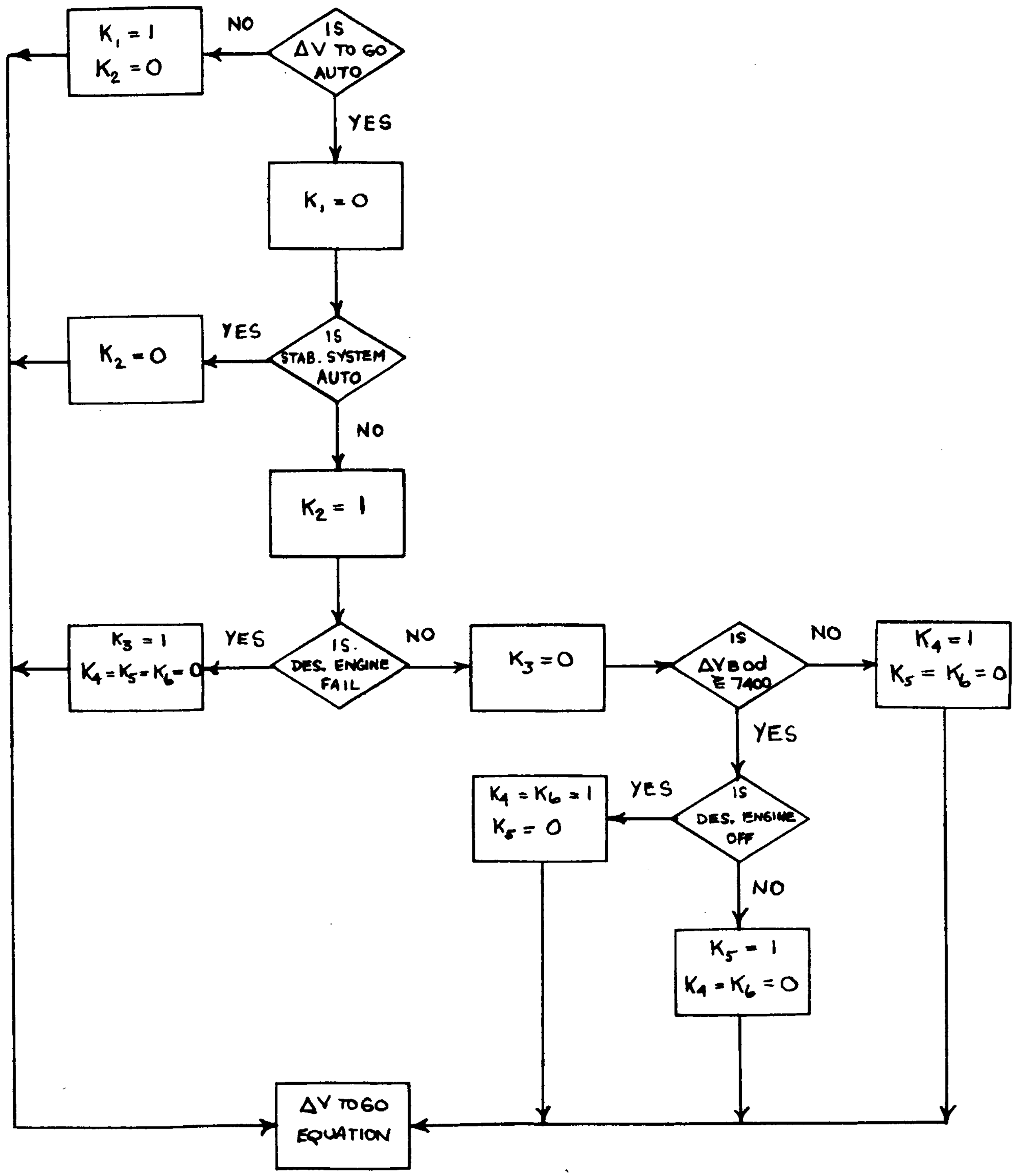
$\Delta V_{USED} = \Delta V_m + \Delta V_a$

K_1 & K_2 = 0 or 1

ΔV_{Boa} and ΔV_{Bod} are on-set constants

The ΔV TO GO logic is illustrated in the following figure:

$V_{TO\ GO}$ LOGIC


IS ΔV TO GO AUTO
NO
YES
K1 = 1
K2 = 0
K1 = 0
IS STAB. SYSTEM AUTO
YES
NO
K2 = 0
K2 = 1
IS. DES. ENGINE FAIL
YES
NO
K3 = 1
K4 = K5 = K6 = 0
K3 = 0
IS ΔVBod ≥ 7400
NO
YES
K4 = 1
K5 = K6 = 0
IS DES. ENGINE OFF
YES
NO
K4 = K6 = 1
K5 = 0
K5 = 1
K4 = K6 = 0
ΔV TO GO EQUATION

APPENDIX B

ABORT SIMULATOR DEFINITION OF SYMBOLS

SYMBOL	DEFINITION	UNITS
B_x	Applied Force Along LEM XB - Body Axis	LBS.
B_y	Applied Force Along LEM YB - Body Axis	LBS.
B_z	Applied Force Along LEM ZB - Body Axis	LBS.
T_j, $j=1...16$	Reaction Jet Thrusts	LBS.
T_m	Descent Engine Thrust	LBS.
T_a	Ascent Engine Thrust	LBS.
$\delta\psi_m$	Descent Engine Gimbal Angle About Body ZB - Axis	RAD.
$\delta\psi_a$	Ascent Engine Thrust Misalignment Angle About Body ZB - Axis	RAD.
$\delta\theta_m$	Descent Engine Gimbal Angle About Body YB - Axis	RAD.
$\delta\theta_a$	Ascent Engine Thrust Misalignment Angle About Body YB - Axis	RAD.
$\Delta\ddot{X}_p$	Acceleration In Inertial XP - Direction Due to Applied Forces	FT/SEC^2
$\Delta\ddot{Y}_p$	Acceleration In Inertial YP - Direction Due to Applied Forces	FT/SEC^2
$\Delta\ddot{Z}_p$	Acceleration In Inertial ZP - Direction Due to Applied Forces	FT/SEC^2
M_T	Instantaneous Total LEM Mass	Slugs
$L_{1,2,3}, M_{1,2,3}, N_{1,2,3}$	Direction Cosines Between Inertial Axes and LEM Body Axes	–
$\dot{L}_{1,2,3}, \dot{M}_{1,2,3}, \dot{N}_{1,2,3}$,	Direction Cosine Rates Between Inertial Axes and LEM Body Axes	–
$\ddot{X}_p$	Total LEM Acceleration In Inertial XP - Direction	FT/SEC^2
$\ddot{Y}_p$	Total LEM Acceleration In Inertial YP - Direction	FT/SEC^2

SYMBOL	DEFINITION	UNITS
$\ddot{Z}_p$	Total LEM Acceleration In Inertial ZP - Direction	FT/SEC^2
g	Gravitational Acceleration - (32.2)	FT/SEC^2
$\dot{X}_p$	LEM Velocity In Inertial XP - Direction	FT/SEC
$\dot{Y}_p$	LEM Velocity In Inertial YP - Direction	FT/SEC
$\dot{Z}_p$	LEM Velocity In Inertial ZP - Direction	FT/SEC
X_p	LEM Distance From Moon Center In Inertial Landing Site Vertical Direction	FT
Y_p	LEM Distance From Moon Center Along Moon Spin Axis In South Direction	FT
Z_p	LEM Distance From Moon Center In Inertial Landing Site Horizontal Direction	FT
R_B	Altitude of the LEM from the Lunar Surface	FT
λ_B	LEM Latitude	RADIANS
δ_B	LEM Longitude	RADIANS
R_p	LEM Radial Distance From Moon Center	FT
V_{n1B}	LEM Inertial Velocity In Local Horizontal Easterly Direction	FT/SEC
V_{n2B}	LEM Inertial Velocity In Local Horizontal Southerly Direction	FT/SEC
V_{r1B}	LEM Inertial Velocity In Local Vertical Direction	FT/SEC
$\dot{\lambda}_B$	LEM Latitude Rate	RAD/SEC
$\dot{\delta}_B$	LEM Longitude Rate	RAD/SEC
X_1	Reaction Jet Moment Arm In XB Direction	FT
$Y_{1,2,3,4}$	Reaction Jet Moment Arms In YB Direction	FT
$Z_{1,2,3,4}$	Reaction Jet Moment Arms In ZB Direction	FT
x_a, y_a, z_a	Ascent Engine Moment Arms	FT
x_m, y_m, z_m	Descent Engine Moment Arms	FT
L	Applied Moment About XB - Axis	FT-LBS.
M	Applied Moment About YB - Axis	FT-LBS.
N	Applied Moment About ZB - Axis	FT-LBS.

SYMBOL	DEFINITION	UNITS
$\dot{p}$	Angular Acceleration About XB - Axis	RAD/SEC^2
$\dot{q}$	Angular Acceleration About YB - Axis	RAD/SEC^2
$\dot{r}$	Angular Acceleration About ZB - Axis	RAD/SEC^2
p	Angular Rate About XB - Axis	RAD/SEC
q	Angular Rate About YB - Axis	RAD/SEC
r	Angular Rate About ZB - Axis	RAD/SEC
I_x	Moment Of Inertia About XB - Axis	$SLUG\text{-}FT^2$
I_y	Moment Of Inertia About YB - Axis	$SLUG\text{-}FT^2$
I_z	Moment Of Inertia About ZB - Axis	$SLUG\text{-}FT^2$
I_{xy}	Product Of Inertia	$SLUG\text{-}FT^2$
I_{yz}	Product Of Inertia	$SLUG\text{-}FT^2$
I_{xz}	Product Of Inertia	$SLUG\text{-}FT^2$
$\dot{M}$	Total Mass Rate	SLUG/SEC
l^2_{DpA}, l^2_{DqrA}	Characteristic Distance For Jet Damping-Ascent Engine	FT^2
l^2_{DpD}, l^2_{DqrD}	Characteristic Distance For Jet Damping-Descent Engine	FT^2
$\ddot{X}_p$	CSM Acceleration In Inertial XP Direction	FT/SEC^2
$\ddot{Z}_p$	CSM Acceleration In Inertial ZP Direction	FT/SEC^2
G	Universal Gravitational Constant	FT/SEC^2
M	Lunar Mass	SLUGS
$\dot{X}_p$	CSM Velocity In Inertial XP Direction	FT/SEC
$\dot{Z}_p$	CSM Velocity In Inertial ZP Direction	FT/SEC
X_p	CSM Distance From Moon's Center In Inertial XP Direction	FT
Z_p	CSM Distance From Moon's Center In Inertial ZP Direction	FT
R_A	CSM Altitude (Constant)	FT

SYMBOL	DEFINITION	UNITS
R_{xp}	Relative Position in Inertial XP-Direction	FT
R_{yp}	Relative Position in Inertial YP-Direction	Ft
R_{zp}	Relative Position in Inertial ZP-Direction	FT
ρ	Range	FT
R_{xB}	Relative Position in XB-Body Axis Direction	FT
R_{yB}	Relative Position in YB-Body Axis Direction	FT
R_{zB}	Relative Position in ZB-Body Axis Direction	FT
A	Azimuth Angle	RAD
E	Elevation Angle	RAD
σ_A	CSM Longitude	RAD
R_{xA}	Relative Position in CSM Local Horizontal Easterly Direction	FT
R_{yA}	Relative Position in CSM Local Horizontal Southerly Direction	FT
R_{zA}	Relative Position in CSM Local Vertical Direction	FT
A_A	Radar Gimbal Angle in CSM Local Horizontal Plane	RAD
E_A	Radar Gimbal Angle in CSM Local Vertical Plane	RAD
U_{xp}	Relative Velocity in Inertial XP Direction	FT/SEC
U_{yp}	Relative Velocity in Inertial YP Direction	FT/SEC
U_{zp}	Relative Velocity in Inertial ZP Direction	FT/SEC
U_{RT}	Total Relative Velocity	FT/SEC
U_{xB}	Relative Velocity in XB-Body Axis Direction	FT/SEC
U_{yB}	Relative Velocity in YB-Body Axis Direction	FT/SEC
U_{zB}	Relative Velocity in ZB-Body Axis Direction	FT/SEC
$\dot{\rho}$	Range Rate	FT/SEC
V_j	Component of Relative Velocity in Line-of-Sight j_1-Direction	FT/SEC

SYMBOL	DEFINITION	UNITS
V_k	Component Of Relative Velocity In Line-Of-Sight k_1 - Direction	FT/SEC
σ_{OA}	Initial CSM Longitude	RAD
$\dot{\sigma}_A$	CSM Longitude Rate	RAD/SEC
t	Time	SEC
θ_c	Command Pitch Angle	RAD
$L_{1,2,3}, M_{1,2,3}, N_{1,2,3}$	Direction Cosines Relating LEM Body Axes To CSM Local Vertical	–
$L^1_{1,2,3}, M^1_{1,2,3}, N^1_{1,2,3}$	Direction Cosines Relating LEM Body Axes To LEM Local Vertical	–
ΔM_{jx+}	Reaction Jet Propellant Burned In + XB - Direction	SLUGS
ΔM_{jy+}	Reaction Jet Propellant Burned In + YB - Direction	SLUGS
ΔM_{jz+}	Reaction Jet Propellant Burned In + ZB - Direction	SLUGS
ΔM_{jx-}	Reaction Jet Propellant Burned In - XB - Direction	SLUGS
ΔM_{jy-}	Reaction Jet Propellant Burned In - YB - Direction	SLUGS
ΔM_{jz-}	Reaction Jet Propellant Burned In - ZB - Direction	SLUGS
ΔM_A	Reaction Jet Propellant Burned In Jet System A	SLUGS
ΔM_m	Reaction Jet Propellant Burned By Descent Engine	SLUGS
ΔM_B	Reaction Jet Propellant Burned By Jet System B	SLUGS
ΔM_a	Reaction Jet Propellant Burned By Ascent Engine	SLUGS
ΔM_T	Total Propellant Burned	SLUGS
ΔV_{TR}	Translational ΔV Consumed	FT/SEC

SYMBOL	DEFINITION	UNITS
I_{SP_j}	Reaction Jet Specific Impulse	SEC
I_{SP_m}	Descent Engine Specific Impulse	SEC
I_{SP_a}	Ascent Engine Specific Impulse	SEC
M_{RCS}	Reaction Jet Fuel	SLUGS
$M_{O\ RCS}$	Initial Reaction Jet Fuel	SLUGS
ΔV_m	ΔV Used by Descent Engine	FT/SEC
ΔV_a	ΔV Used by Ascent Engine	FT/SEC
ΔV_{Tm}	ΔV Remaining in Descent Stage	FT/SEC
ΔV_{Ta}	ΔV Remaining in Ascent Stage	FT/SEC
M_{OT1}	Initial Mass - Descent Stage	SLUGS
M_{OT2}	Initial Mass - Ascent Stage	SLUGS
ΔV_{ABm}	ΔV Required for Abort Before Staging	FT/SEC
ΔV_{ABa}	ΔV Required for Abort After Staging	FT/SEC
$I_{XO1}, I_{YO1}, I_{ZO1}$	Initial Moments of Inertia for LEM Separation Weight	$SLUG\text{-}FT^2$
$I_{XYO1}, I_{YZO1}, I_{XZO1}$	Initial Products of Inertia for LEM Separation Weight	$SLUG\text{-}FT^2$
$I_{XO2}, I_{YO2}, I_{ZO2}$	Initial Moments of Inertia after Staging Ascent Engine	$SLUG\text{-}FT^2$
$I_{XYO2}, I_{YZO2}, I_{XZO2}$	Initial Products of Inertia after Staging Ascent Engine	$SLUG\text{-}FT^2$
$\left(\frac{dI_x}{dM_T}\right)_1 \left(\frac{dI_y}{dM_T}\right)_1 \left(\frac{dI_z}{dM_T}\right)_1$	Derivative of Moments of Inertia with Respect to Total Mass Change-Descent Stage	FT^2
$\left(\frac{dI_{xy}}{dM_T}\right)_1 \left(\frac{dI_{yz}}{dM_T}\right)_1 \left(\frac{dI_{xz}}{dM_T}\right)_1$	Derivative of Products of Inertia with Respect to Total Mass Change-Descent Stage	FT^2
$\left(\frac{dI_x}{dM_T}\right)_2 \left(\frac{dI_y}{dM_T}\right)_2 \left(\frac{dI_z}{dM_T}\right)_2$	Derivative of Moments of Inertia with Respect to Total Mass Change-Ascent Stage	FT^2
$\left(\frac{dI_{xy}}{dM_T}\right)_2 \left(\frac{dI_{yz}}{dM_T}\right)_2 \left(\frac{dI_{xz}}{dM_T}\right)_2$	Derivative of Products of Inertia WRT Total Mass Change-Ascent Stage	FT^2

SYMBOL	DEFINITION	UNITS
X_{CG}, Y_{CG}, Z_{CG}	Distances Between Center of Gravity and Reference Point	FT
X_{CGO_1}, Y_{CGO_1}, Z_{CGO_1}	Initial Position of CG at Separation	FT
X_{CGO_2}, Y_{CGO_2}, Z_{CGO_2}	Initial Position of CG After Staging	FT
$\left(\frac{dX_{CG}}{dM_T}\right)_1 \left(\frac{dY_{CG}}{dM_T}\right)_1 \left(\frac{dZ_{CG}}{dM_T}\right)_1$	Change in CG Position WRT Change in Mass for Descent Stage	FT/SLUG
$\left(\frac{dX_{CG}}{dM_T}\right)_2 \left(\frac{dY_{CG}}{dM_T}\right)_2 \left(\frac{dZ_{CG}}{dM_T}\right)_2$	Change in CG Position WRT Change in Mass for Ascent Stage	FT/SLUG
x_m, y_m, z_m	Descent Engine Moment Arms	FT
X_{TM}, Y_{TM}, Z_{TM}	Distance Between Fixed Reference Point and Descent Engine Gimbal Axis Point	FT
x_a, y_a, z_a	Ascent Engine Moment Arms	FT
X_{Ta}, Y_{Ta}, Z_{Ta}	Distance Between Fixed Reference Point and Ascent Engine Nozzle	FT
X_{T_1}	Distance Between Fixed Reference Point and RCS Nozzle in X_B Direction	FT
$Y_{T_{1,2,3,4}}$	Distance Between Fixed Reference Point and RCS Nozzles in Y_B Direction	FT
$Z_{T_{1,2,3,4}}$	Distance Between Fixed Reference Point and RCS Nozzles in Z_B Direction	FT
V_X	LEM Velocity in the X_B Body Axis Direction	FT/SEC
θ	Rotation About the Y_B Body Axis WRT Inertial Space (Pitch)	RAD
ψ	Rotation About the Z_B Body Axis WRT Inertial Space (Roll)	RAD
ϕ	Rotation About the X_B Body Axis WRT Inertial Space (Yaw)	RAD
θ_A	Rotation About the Y_B Body Axis WRT CSM Local Vertical (Pitch)	RAD
ψ_A	Rotation About the Z_B Body Axis WRT CSM Local Vertical (Roll)	RAD
ϕ_A	Rotation About the X_B Body Axis WRT CSM Local Vertical (Yaw)	RAD

SYMBOL	DEFINITION	UNITS
θ_T	Rotation about the Y Thrust Axis WRT Inertial Space (Pitch)	RAD
ψ_T	Rotation About the Z Thrust Axis WRT Inertial Space (Roll)	RAD
ϕ_T	Rotation about the X Thrust Axis WRT Inertial Space (Yaw)	RAD

Appendix C

Range of Variables

Table C-1 contains the ranges of variables for the Grumman Abort Simulation.

TABLE C
RANGE OF VARIABLES FOR GAEC ABORT SIMULATION

Symbol	Range	Units	Symbol	Range	Units
B_x	-400 - +10900	lbs	λ_B	-0.03 - +0.03	RAD
B_y	-200 - +1300	lbs	σ_B	-0.18 - +5.0	RAD
B_z	-200 - +1300	lbs	R_p	5.70209973×10^6 - $+1.6\times10^6$	FT
T_j, j=1....16	0 - 100	lbs	V_{n1B}	0 - +5,700	FT/SEC
T_m	0 - 10,500	lbs	V_{n2B}	-100 - +100	FT/SEC
T_a	0 - 3,500	lbs	V_{r1B}	-300 - +400	FT/SEC
$\delta\psi_m, \delta\theta_m$	-0.10473 - +0.10473	RAD	$\dot{\lambda}_B$	-2×10^{-4} - $+2\times10^{-4}$	RAD/SEC
$\delta\psi_a, \delta\theta_a$	-0.06 - +0.06	RAD	$\dot{\sigma}_B$	0 - $+1\times10^{-3}$	RAD/SEC
$\Delta\ddot{x}_p$	-30 - + 30	FT/SEC^2	x_1	0 - +4.5	FT
$\Delta\ddot{y}_p$	-3 - + 3	FT/SEC^2	y_1	+4.9 - +5.1	FT
$\Delta\ddot{z}_p$	-30 - + 30	FT/SEC^2	y_2	-4.9 - -5.1	FT
M_T	110 - 797.61	SLUGS	y_3	+5.4 - +5.6	FT
$\ell_{1,2,3}, m_{1,2,3}, n_{1,2,3}$	-1 - + 1	-	y_4	-5.4 - -5.6	FT
$\dot{\ell}_{1,2,3}, \dot{m}_{1,2,3}, \dot{n}_{1,2,3}$	-0.35-+0.35	1/SEC	z_1	+5.2 - +5.8	FT
$\ddot{x}_p$	-35 - +25	FT/SEC^2	z_2	-5.2 - -5.8	FT
$\ddot{y}_p$	- 3 - + 3	FT/SEC^2	z_3	+4.7 - +5.3	FT
$\ddot{z}_p$	-30 - +30	FT/SEC^2	z_4	-4.7 - -5.3	FT
g	32.2	FT/SEC^2	x_a	-1.0 - 0	FT
$\dot{x}_p$	-5,700 - +5,700	FT/SEC	y_a	-0.009-+0.017	FT
$\dot{y}_p$	-100 - + 100	FT/SEC	z_a	-0.25- - 0.05	FT
$\dot{z}_p$	-5,700-+5,700	FT/SEC	x_m	-4.5 - - 1.8	FT
x_p	-1.1×10^6-$+1.1\times10^6$	FT	y_m	0.009-+0.017	FT
y_p	-2×10^4-$+2\times10^4$	FT	z_m	-0.25- -0.05	FT
z_p	-1.1×10^6-$+1.1\times10^6$	FT	L	-2,000-+2,000	FT-lbs
R_B	0-$+1.1\times10^6$	FT	M, N	-7,000-+7,000	FT-lbs
R_M	5.70209973×10^6	FT	R_{xp}	$\pm 1.5 \times 10^6$	FT
$\dot{p}$	± 1.0	RAD/SEC^2	R_{yp}	$\pm 2 \times 10^4$	FT
$\dot{q}$	± 1.0	RAD/SEC^2	R_{zp}	$\pm 1.5 \times 10^6$	FT
$\dot{r}$	± 1.0	RAD/SEC^2	ρ	1×10^5-$+1.5\times10^6$	FT
p	±0.35	RAD/SEC	R_{xB}	$\pm1.5\times10^6$	FT
q	±0.35	RAD/SEC	R_{yB}	$\pm1.5\times10^6$	FT
r	±0.35	RAD/SEC	R_{zB}	$\pm1.5\times10^6$	FT
I_x	2,300-+17,504	$SLUG\text{-}FT^2$	A	± 4.7	RAD
I_y	2,100-+19,766	$SLUG\text{-}FT^2$	E	± 6.3	RAD
I_z	1,100-+18,989	$SLUG\text{-}FT^2$	σ_A	-0.326351-+5.0	RAD
I_{xy}	-11 -+19	$SLUG\text{-}FT^2$	R_{xA}	$\pm 1.5 \times10^6$	FT

Symbol	Range	Units	Symbol	Range	Units
I_{yz}	-191- -80	SLUG-FT^2	R_{yA}	± 2 x10^4	FT
I_{xz}	52 - +232	SLUG-FT^2	R_{zA}	±1.5x10^6	FT
$\dot{M}$	0 - +0.95	SLUG/SEC	A_A	±6.3	RAD
l_{Dpa}	3.88	FT	E_A	± 6.3	RAD
l^2_{DqrA}	18.8	FT^2	U_{xp}	±500	FT/SEC
l_{DpD}	3.91	FT	U_{yp}	±100	FT/SEC
l^2_{DqrD}	30.4	FT^2	U_{zp}	-500-+5700	FT/SEC
$\ddot{X}_{pA}$	-5.3134-+5.3134	FT/SEC^2	U_{RT}	-500-+5700	FT/SEC
$\ddot{Z}_{pA}$	-5.3134-+5.3134	FT/SEC^2	U_{xB}	±5700	FT/SEC
$\dot{X}_{pA}$	±5284.69883	FT/SEC	U_{yB}	±5700	FT/SEC
$\dot{Z}_{pA}$	±5284.69883	FT/SEC	U_{zB}	±5700	FT/SEC
X_{pA}	±6.1881897x10^6	FT	$\dot{\rho}$	±500	FT/SEC
Z_{pA}	±6.1881897x10^6	FT	V_j	±500	FT/SEC
R_A	6.1881897x10^6	FT	V_k	±500	FT/SEC
σ_{oA}	-0.326351	RAD	ΔV_{Ta}	0-+7,200	FT/SEC
$\dot{\sigma}_A$	8.53984283x10^{-4}	RAD/SEC	$\Delta V_{RESET\ 1}$	0-+7,000	FT/SEC
t	0-9,999	SEC	$\Delta V_{RESET\ 2}$	0-+7,000	FT/SEC
E_p	-1 -+1	-	M_{T01}	798.35	SLUGS
E_q	-1-+1	-	M_{T02}	264.45	SLUGS
E_r	-1-+1	-	ΔV_{ABm}	0-+7,000	FT/SEC
Θ_c	-6.3-+6.3	RAD	V_{ABa}	0-+7,000	FT/SEC
$L_{1,2,3}, M_{1,2,3}\ N_{1,2,3}$	-1-+1	-	I_{x01}	17,504	SLUG-FT^2
$L'_{1,2,3}, M'_{1,2,3}, N'_{1,2,3}$	-1-+1	-	I_{y01}	19,766	SLUG-FT^2
$\Delta M_{jx+}, \Delta M_{jx-}$	0-+12	SLUGS	I_{z01}	18,989	SLUG-FT^2
$\Delta M_{jy+}, \Delta M_{jy-}$	0-+12	SLUGS	I_{xy01}	19	SLUG-FT^2
$\Delta M_{jz+}, \Delta M_{jz-}$	0-+12	SLUGS	I_{yz01}	-191	SLUG-FT^2
$\Delta M_B, \Delta M_B$	0-+16	SLUGS	I_{xz01}	53	SLUG-FT^2
ΔM_m	0-+460	SLUGS	I_{x02}	4,738	SLUG-FT^2
ΔM_a	0-+140	SLUGS	I_{y02}	2,935	SLUG-FT^2
ΔM_T	0-+460	SLUGS	I_{z02}	4,083	SLUG-FT^2
ΔV_{TR}	0-+14,900	FT/SEC	I_{xy02}	-11	SLUG-FT^2
I_{SPj}	300	SEC	I_{yz02}	-84	SLUG-FT^2
I_{SPm}	305	SEC	I_{xz02}	232	SLUG-FT^2
I_{SPa}	306.3	SEC	$(dI_x/dM_T)_1$	22.0278	FT^2
M_{RCS}	0-+16	SLUGS	$(dI_y/dM_T)_1$	24.25805	FT^2
Mo_{RCS}	16	SLUGS	$(dI_z/dM_T)_1$	19.701915	FT^2

Symbol	Range	Units	Symbol	Range	Units
ΔV_m	0-+8,000	FT/SEC	$(dI_{xy}/dM_T)1$	-0.04134	FT^2
ΔV_a	0-+7,200	FT/SEC	$(dI_{yz}/dM_T)1$	$-2.1758+10^{-3}$	FT^2
ΔV_{Tm}	0-+8,000	FT/SEC	$(dI_{xz}/dM_T)1$	-0.202350	FT^2
$(dI_x/dM_T)2$	15.66941	FT^2	Y_{Ta}	0	FT
$(dI_y/dM_T)2$	4.87658	FT^2	Z_{Ta}	0	FT
$(dI_z/dM_T)2$	19.491399	FT^2	X_{T1}	20.5	FT
$(dI_{xy}/dM_T)2$	0	FT^2	Y_{T1}	+5.0	FT
$(dI_{yz}/dM_T)2$	0	FT^2	Y_{T2}	-5.0	FT
$(dI_{xz}/dM_T)2$	0.022438	FT^2	Y_{T3}	+5.5	FT
X_{CG}	13.59160-20.45	FT	Y_{T4}	-5.5	FT
Y_{CG}	-0.008333-+.01667	FT	Z_{T1}	+5.5	FT
Z_{CG}	-0.04167-+0.24167	FT	Z_{T2}	-5.5	FT
X_{CG01}	15.59166	FT	Z_{T3}	+5.0	FT
Y_{CG01}	-0.008333	FT	Z_{T4}	-5.0	FT
Z_{CG01}	-0.0416007	FT	θ	± 6.3	RAD
X_{CG02}	20.45	FT	ψ	± 1.5	RAD
Y_{CG02}	0.009333	FT	ϕ	± 6.3	RAD
Z_{CG02}	-0.125	FT	θ_A	± 6.3	RAD
$(dX_{CG}/dM_T)1$	$-5.7114880+10^{-3}$	FT/SLUG	ψ_A	± 1.5	RAD
$(dY_{CG}/dM_T)1$	$1.813+10^{-5}$	FT/SLUG	ϕ_A	± 6.3	RAD
$(dZ_{CG}/dM_T)1$	$1.087898+10^{-4}$	FT/SLUG	$M_M = G_M$	$1.7282388+10^{14}$	FT^3/SEC
$(dX_{CG}/dM_T)2$	$6.2328+10^{-4}$	FT/SLUG	$l_{1,2,3_T}\, m_{1,2,3_T}\, n_{1,2,3_T}$	± 1	-
$(dY_{CG}/dM_T)2$	-6.2328×10^{-5}	FT/SLUG	θ_T	± 6.3	RAD
$(dZ_{CG}/dM_T)2$	8.7262528×10^{-4}	FT/SLUG	ψ_T	± 1.5	RAD
X_{Tm}	13.79166	FT	ϕ_T	± 6.3	RAD
Y_{Tm}	0	FT			
Z_{Tm}	0	FT			
X_{Ta}	19.58333				

APPENDIX D

PITCH PROGRAM DATA

GAEC LMO-500-101 forms the content of this Appendix and contains all Pitch Program Data.

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GROUP 4 DOCUMENT
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DECLASSIFIED AFTER 12 YEARS

GRUMMAN AIRCRAFT ENGINEERING CORPORATION

LEM ENGINEERING MEMORANDUM

LMO-500-101
11 October 1963
693-041

From: L. Weber and D. McCabe, Dynamics - Systems Engineering, x1461

To:

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Subject: FIRM ABORT TRAJECTORY DATA FOR THE LINE-OF-SIGHT ABORT TECHNIQUE AND CORRECTED DATA FOR INERTIAL ABORT TECHNIQUE

Reference: (1) LEM Engineering Memo LMO-500-95, "Data for LTV Abort Simulation Program", dated 27 September 1963.

SUMMARY

This memo presents the firm abort trajectory data for the line-of-sight abort technique for use in the LTV abort simulation. In addition, corrected data for the inertial abort technique, are included.

LINE-OF-SIGHT TECHNIQUE ABORTS

Reference (1) presented firm inertial and estimated line-of-sight abort procedures and trajectory data for the LTV abort simulation program. Tables (1) and (2) of this memo present the firm line-of-sight trajectory data. In addition, two modifications of the LOS abort procedure discussed in Reference (1) will be defined below:

(1) Descent/Ascent Engine:

The abort procedure, when the descent/ascent engines are to be used is the same as denoted in reference (1) except that the descent engine throttle

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is set, for all abort points, to produce a thrust of 6,000 lbs. This thrust level is held constant until either descent engine burnout or staging occurs. A lower thrust level for the descent/ascent cases, for early aborts, is necessary to satisfy radar look angle and desired burnout altitude constraints.

(2) Early Aborts:

In early abort cases, an 18-second roll through an angle of 180° is necessary, prior to the pitch rotation, to satisfy radar look angle requirements during abort ascent. During this 18-second period, the inertial pitch angle is held constant. In more detail, the procedure for the ascent and descent/ascent engine abort cases are noted below:

(a) Ascent Engine:

At abort, the vehicle is rolled 180 deg. (from belly-up to belly down) at 10 deg/sec. roll rate. The vehicle is coasting during this 18-second roll period. The vehicle is then rotated, at the 10 deg/sec. inertial pitch rate, until the required LOS elevation angle is achieved (E_1). The ascent engine is then lit and the procedure continues as discussed in reference (1).

(b) Descent/Ascent Engine:

The same 18-second roll procedure is followed. However, during this time the descent engine is operating at 10,500 lb. thrust. At the completion of this roll maneuver, the descent engine is then throttled to a thrust of 6,000 lbs, and the pitch maneuver to E_1 performed. The remainder of the procedure is the same as that discussed in reference (1).

INERTIAL TECHNIQUE ABORTS

Certain errors were discovered in Tables 1 and 2 of reference (1). These errors were corrected and are presented in Tables 3 and 4 of the present memo. For ease of determining which values were changed, all changes to Tables 1 and 2 of reference (1) are underlined.

LW:jlp

[redacted]

TABLE - 1

FIRM LINE OF SIGHT PITCH PROGRAM ABORT PROCEDURE AND TRAJECTORY DATA

ASCENT ENGINE ONLY

ABORT POINT	1	2	3	4	5
A. ABORT PROCEDURE TO BURNOUT:					
(1) AT ABORT (POINT 0*):					
t_a, SEC.	40.	160.	280.	390.4	510.47
θ_a, DEG.	187.8624	176.7033	165.5671	130.1212	90.1230
ΔV USED$_a$, FT/SEC	541.2991	2371.8270	4623.5296	6003.8579	6661.8352
ΔV TO GO$_a$, FT/SEC	2957.3020	5349.8072	5100.9615	5775.4245	6102.7359
(2) AFTER ROLLING 180°:					
t', SEC.	58.	178.	NO ROLL	NO ROLL	NO ROLL
θ', DEG.	187.8624	176.7033	—	—	—
(3) AFTER ERECTION** (POINT 1):					
E_1, DEG.	-53.4881	-41.6692	132.9751	93.4438	32.5160
(4) AT POINT 2:					
t_1, SEC	177.6615	419.5407	306.8401	398.1184	551.3334
ΔV USED$_1$, FT/SEC.	2007.1260	5937.1914	4882.7197	6011.8949	7217.8784
ΔV TO GO$_1$, FT/SEC.	1491.4751	1784.4428	4841.7774	5767.3861	5546.6927
(5) AT POINT 3**:					
E_2, DEG.	141.6811	150.9341	154.2713	107.6388	92.5678
(6) AT BURNOUT:					
$t_{B.O.}$, SEC.	267.5870	327.2667	588.2986	726.0810	854.2346
ΔV USED$_{B.O.}$, FT/SEC	3498.6011	7721.6342	9724.4971	11779.2824	12764.5711
ΔV TO GO, FT/SEC	0.0	0.0	0.0	0.0	0.0
B. ABORT PROCEDURE FOR TRANSFER ORBIT INSERTION:					
(1) AT INITIATION OF THRUST:					
t_T, SEC	300.5129		2255.9750	1055.3086	2346.8977
ρ_T, FT	1295485.		789587.	884886.	884598.
E_T, DEG	-79.6293		-62.5772	116.2086	116.1692
ΔV USED$_T$, FT/SEC	3498.6011		9724.4971	11779.2824	12764.5711
ΔV TO GO$_T$, FT/SEC	190.0000		173.5000	110.0000	110.0000
(2) AT TRANSFER ORBIT INSERTION:					
t_i, SEC	312.4229		2263.4550	1059.8786	2351.4477
ΔV USED$_i$, FT/SEC	3688.6011		9897.9971	11889.2824	12874.5711
ΔV TO GO$_i$, FT/SEC	0.0		0.0	0.0	0.0
C. TRAJECTORY PARAMETERS:					
(1) AT BURNOUT:					
h_{BO}, FT	100,000.	75,000.	75,000.	50,000.	50,000.
V_{BO}, FT/SEC	5457.695	5469.483	5469.483	5481.328	5481.328
γ_{BO}, DEG	0.0	0.0	0.0	0.0	0.0
$\sigma_{B\,BO}$, DEG	4.8236	14.3187	8.0107	7.7809	7.4586
$\sigma_{R\,BO}$, DEG	10.2108	8.0763	-2.2945	-9.2660	-15.8587
(2) AT INITIATION OF TRANSFER ORBIT INSERTION THRUST:					
h_T, FT	100,000.		75,000.	50,000.	50,000.
V_T, FT/SEC	5457.695		5469.483	5481.328	5481.328
γ_T, DEG	0.0		0.0	0.0	0.0
σ_{BT}, DEG.	6.5981		97.2312	25.7564	88.9566
σ_{RT}, DEG	11.8456		6.4634	-7.3995	-7.3963

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* SEE NEXT PAGE FOR DESIGNATION OF CRITICAL POINTS

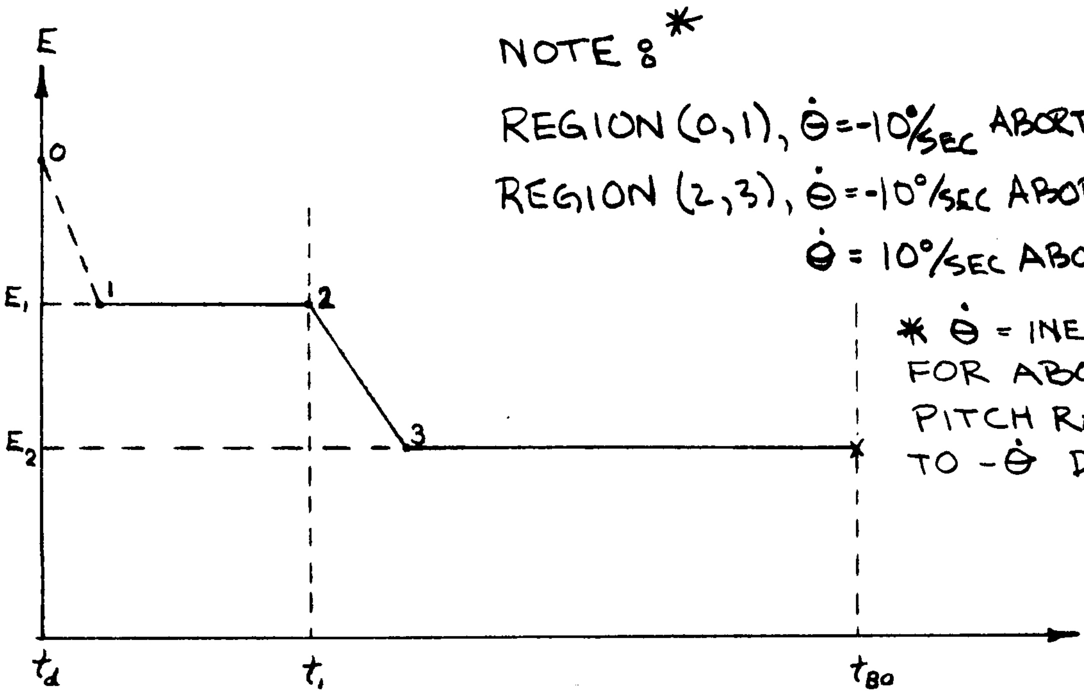
** SEE NEXT PAGE FOR DIRECTION OF PITCH ROTATION TO REQUIRED E.

TABLE-1

FIRM LINE OF SIGHT PITCH PROGRAM ABORT PROCEDURE AND TRAJECTORY DATA -(CONT.)

ASCENT ENGINE ONLY

ABORT POINT	1	2	3	4	5
C. TRAJECTORY PARAMETERS (CONT.):					
(3) AT TRANSFER ORBIT INSERTION:					
h_i, FT	99992.7		74996.3	49998.6	49998.8
V_i, FT/SEC	5647.685		5642.978	5591.326	5591.326
γ_i, DEG.	.0008		-.0016	-.0015	-.0005
σ_{B_i}, DEG.	7.2508		97.6434	26.0083	89.2015
σ_{R_i}, DEG.	11.9158		6.5096	-7.3711	-7.3680
(4) AT APOCYNTHION OF TRANSFER ORBIT: h_a, FT	984,592		870,671	535,865	535,865
(5) AT RENDEZVOUS:					
t_R, SEC.	6140.2457		7800.9320	5323.4506	6614.6017
σ_{B_R}, DEG	280.4933		362.0852	241.9981	305.1737
(6) DURING TRANSFER					
$\sigma_{B_R} - \sigma_{B_i}$, DEG.	273.2425		264.4418	215.9898	215.9662



DESIGNATION OF CRITICAL POINTS FOR PITCH PROGRAM

NOTE:*

REGION (0,1), $\dot{\theta} = -10°/\text{SEC}$ ABORT PTS. 1,2,3,4,5

REGION (2,3), $\dot{\theta} = -10°/\text{SEC}$ ABORT PTS. 3,4,5

$\dot{\theta} = 10°/\text{SEC}$ ABORT PTS. 1,2

* $\dot{\theta}$ = INERTIAL PITCH RATE. FOR ABORT PTS 1, & 2, PILOT PITCH RATE, q, IS EQUIVALENT TO $-\dot{\theta}$ DUE TO 180° ROLL.

TABLE - 2

FIRM LINE OF SIGHT PITCH PROGRAM ABORT PROCEDURE AND TRAJECTORY DATA

DESCENT / ASCENT ENGINE

ABORT POINT	1	2	3	4	5
C ABORT PROCEDURE TO BURNOUT					
(1) AT BURNOUT (POINT 0 *):					
t_d, SEC.	40.	160.			510.47
θ_d, DEG.	187.8624	176.7033			90.1230
ΔV USED$_d$, FT/SEC.	541.2991	2371.8270			6661.8352
ΔV TO GO$_d$, FT/SEC.	2454.7660	5028.1730			738.1648
(2) AFTER ROLLING 180°					
t, SEC θ, DEG	58. 187.8624	178. 176.7033			NO ROLL
(3) AFTER ERECTION (POINT 1): **					
E_1, DEG	-52.1014	-37.0288			32.5160
(4) AT POINT 2:					
t_1, SEC.	210.3732	483.2069			542.9562
ΔV USED$_1$, FT/SEC	2122.2561	6276.9701			7155.1457
ΔV TO GO$_1$, FT/SEC	873.8090	1123.0299			244.8543
(5) AT POINT 3: **					
E_2, DEG	155.7389	162.2161			91.6712
(6) AT BURNOUT:					
t_{BO}, SEC	298.7746	566.6708			869.234
ΔV USED$_{BO}$, FT/SEC	2996.0651	7519.8409			12723.0287
ΔV TO GO$_{BO}$, FT/SEC	0.0	0.0			0.0
B ABORT PROCEDURE STAGING AND THRUST DATA:					
(1) STAGING DATA:					
ΔV USED, AT STAGING, FT/SEC.	NO STAGING	7400.0			7400.0
ΔV TO GO, BEFORE STAGING, FT/SEC	–	0.0			0.0
t_s, STAGING TIME, SEC	–	557.7009			558.4845
ΔV TO GO$_s$, AFTER STAGING, FT/SEC	–	119.8409			5323.0287
(2) THRUST DATA					
T_d, THRUST AT ABORT, LBS.	10,500.	10,500.			6000.
t_c, TIME TO CUT THRUST, SEC.	58	178.			510.47
ΔV USED, AT THRUST CUT BACK, FT/SEC	———	———			6661.8352
T_c, VALUE OF CUT BACK THRUST, LBS.	6000.	6000.			6000.
C ABORT PROCEDURE FOR TRANSFER ORBIT INSERTION					
(1) AT INITIATION OF THRUST:					
t_T, SEC.	751.9389				2389.6082
S_T, FT	1,295,624.				884,624.
E_T, DEG	-79.6318				+115.905
ΔV USED$_T$, FT/SEC	2996.0651				12723.0287
ΔV TO GO$_T$, FT/SEC	190.0000				110.0000
T_T, THRUST, LBS.	10,500.				3,500.
(2) AT TRANSFER ORBIT INSERTION:					
t_i, SEC	763.1489				2394.6282
ΔV USED$_i$, FT/SEC	3186.0651				12833.6287
ΔV TO GO$_i$, FT/SEC	0.0				0.0
T_i, THRUST, LBS	10,500				3500.

* SEE NEXT PAGE FOR DESIGNATION OF CRITICAL POINTS

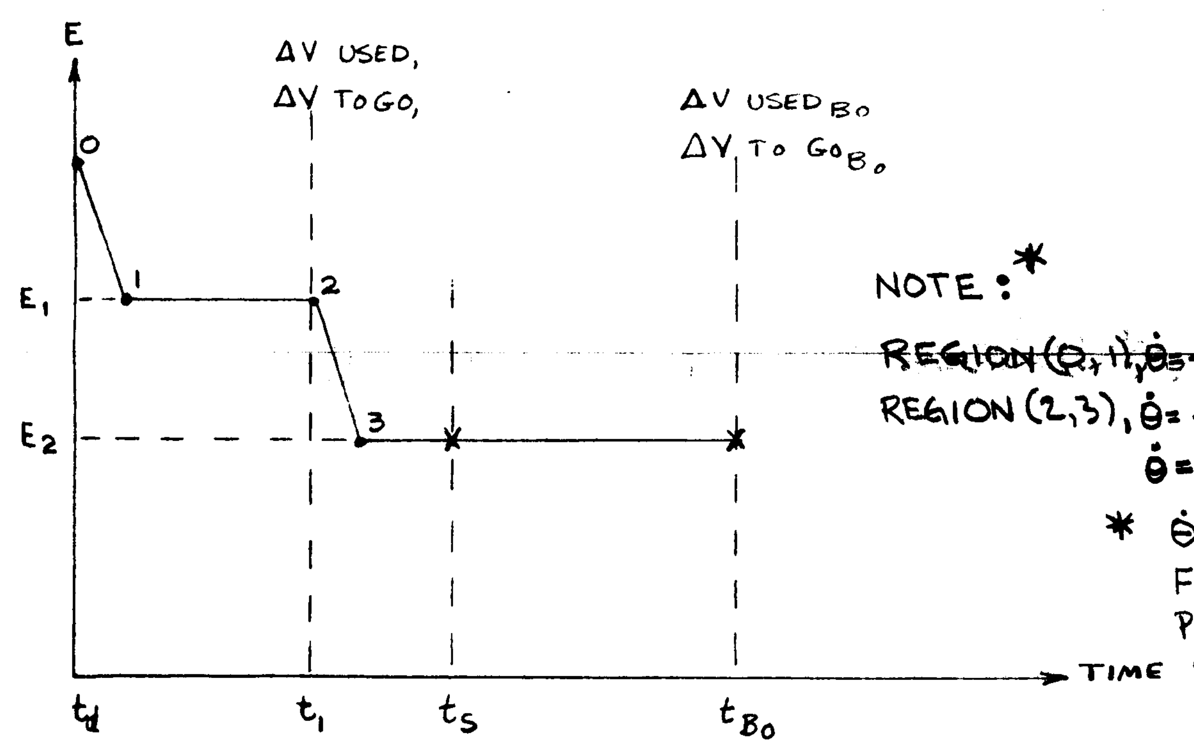
** SEE NEXT PAGE FOR DIRECTION OF PITCH ROTATION TO REQUIRED E.

[redacted]

TABLE - 2 (CONT)
FIRM LINE OF SIGHT PITCH PROGRAM
ABORT PROCEDURE AND TRAJECTORY DATA

DESCENT/ASCENT ENGINE

ABORT POINT	1	2	3	4	5
D TRAJECTORY PARAMETERS:					
(1) AT BURNOUT:					
h_{BO}, FT.	100,000.	75,000.			50,000.
V_{BO}, FT/SEC	5457.695	5469.483			5481.328
γ_{BO}, DEG.	0.0	0.0			0.0
$\sigma_{B_{BO}}$, DEG.	5.7847	15.8627			8.0352
$\sigma_{R_{BO}}$, DEG.	9.5970	6.6158			-16.0161
(2) AT INITIATION OF TRANSFER ORBIT INSERTION THRUST:					
h_T, FT.	100,000.				50,000.
V_T, FT/SEC.	5457.695				5481.328
γ_T, DEG	0.0				0.0
σ_{B_T}, DEG	30.2079				91.0462
σ_{R_T}, DEG.	11.8470				-7.3966
(3) AT TRANSFER ORBIT INSERTION					
h_i, FT.	99,992.5				49,998.8
V_i, FT/SEC.	5647.689				5591.327
γ_i, DEG	-.0011				.0001
σ_{B_i}, DEG.	30.8228				91.3173
σ_{R_i}, DEG	11.9132				-7.3661
(4) AT APOCYNTHION OF TRANSFER ORBIT:					
h_a, FT.	984,664.				535,865.
(5) AT RENDEZVOUS					
t_R, SEC.	6591.5289				6652.5002
σ_{B_R}, DEG	304.0949				307.2690
(6) DURING TRANSFER					
$\sigma_{B_R} - \sigma_{B_i}$, DEG	273.2721				215.9517



NOTE:*
~~REGION (0, 1), $\dot{\theta}$ = -10°/SEC ABORT PTS 1,2,5~~
REGION (2,3), $\dot{\theta}$ = -10°/SEC ABORT PT. 5
$\dot{\theta}$ = 10°/SEC ABORT PTS 1,2
* $\dot{\theta}$ = INERTIAL PITCH RATE. FOR ABORT PTS 1,&2, PILOT PITCH RATE, $\dot{\theta}$, IS EQUIVALENT TO $-\dot{\theta}$ DUE TO 180° ROLL.

DESIGNATION OF CRITICAL POINTS FOR PITCH PROGRAM

[redacted]

[redacted]

TABLE 3

FIRM INERTIAL PITCH PROGRAM
ABORT PROCEDURE AND TRAJECTORY DATA

ASCENT ENGINE ONLY

ABORT POINT	1	2	3	4	5
A. ABORT PROCEDURE TO BURNOUT:					
(1) AT ABORT (POINT 0)****:					
t_a, SEC.	40	160	280	390.4	510.47
θ_a, DEG.	187.8624	176.7033	165.5671	130.1212	90.1230
ΔV_{USED_a}, FT/SEC	541.2991	2371.8270	4623.5296	6003.8579	6661.8352
$\Delta V_{TO\ GO_a}$, FT/SEC	1751.2560	2915.6250	5442.6200	5840.6920	5883.1210
(2) AFTER ERECTION (POINT 1):					
θ, DEG.	90.0000	90.0000	90.0000	90.0000	90.0000
(3) AT POINT 2:					
t_1, SEC.	106.2405	119.8905	355.0462	407.7836	520.9010
ΔV_{USED_1}, FT/SEC	1318.2040	3074.6352	5560.0837	6187.1913	6800.7929
$\Delta V_{TO\ GO_1}$, FT/SEC	974.3511	2212.8168	4506.0659	5657.3586	5744.1633
(4) AT POINT 3:					
θ_1, DEG.	-73.9165	1.4281	21.7703	26.2927	26.4924
(5) AT POINT 4:					
t_2, SEC.	154.9321	273.8163	392.6477	499.5581	615.6287
ΔV_{USED_2}, FT/SEC	2041.2991	3871.8270	6123.5296	7503.8579	8161.8352
$\Delta V_{TO\ GO_2}$, FT/SEC	251.2560	1415.6250	3942.6200	4340.6920	4383.1210
(6) AT POINT 5:					
θ_2, DEG.	-85.9165	-10.5719	9.7703	14.2927	14.4924
(7) AT POINT 6:					
t_3, SEC.	—	—	482.9488	589.8591	705.9294
ΔV_{USED_3}, FT/SEC	—	—	7623.5296	9003.8579	9661.8352
$\Delta V_{TO\ GO_3}$, FT/SEC	—	—	2442.6200	2840.6920	2883.1210
(8) AT POINT 7:					
θ_3, DEG.	—	—	-2.2297	2.2927	2.4924
(9) AT POINT 8:					
t_4, SEC.	—	—	560.4997	667.4101	783.4804
ΔV_{USED_4}, FT/SEC	—	—	9123.5296	10503.8579	11161.8352
$\Delta V_{TO\ GO_4}$, FT/SEC.	—	—	942.6200	1340.6920	1383.1210
(10) AT POINT 9:					
θ_4, DEG.	—	—	-13.2297	-8.7073	-8.5076
(11) AT BURNOUT:					
t_{BO}, SEC.	171.03	359.39	603.5800	727.4100	845.2500
$\Delta V_{USED_{BO}}$, FT/SEC	2292.5551	5287.452	10066.1496	11844.5499	12544.9562
$\Delta V_{TO\ GO_{BO}}$, FT/SEC	0.0	0.0	0.0	0.0	0.0
B. ABORT PROCEDURE FOR TRANSFER ORBIT INSERTION:					
(1) AT INITIATION OF THRUST:					
t_T, SEC.	801.1079	496.8234	2398.7562	1056.8865	2195.6098
θ_T, DEG.	-33.4363	-13.0072	-105.9464	-26.2137	-81.8842
ρ_T, FT.	1,295,485	788,615	789,587	884,886	884,598
ΔV_{USED_T}, FT/SEC.	2292.5551	5287.452	1066.1496	11844.5499	12544.9562
$\Delta V_{TO\ GO_T}$, FT/SEC.	190.0000	173.5000	173.5000	110.0000	110.0000
(2) AT TRANSFER ORBIT INSERTION:					
t_i, SEC.	813.0179	506.4834	2406.2362	1061.4565	2200.1598
ΔV_{USED_i}, FT/SEC.	2482.5551	5460.952	10239.6496	11854.5499	12654.9562
$\Delta V_{TO\ GO_i}$	0.0	0.0	0.0	0.0	0.0

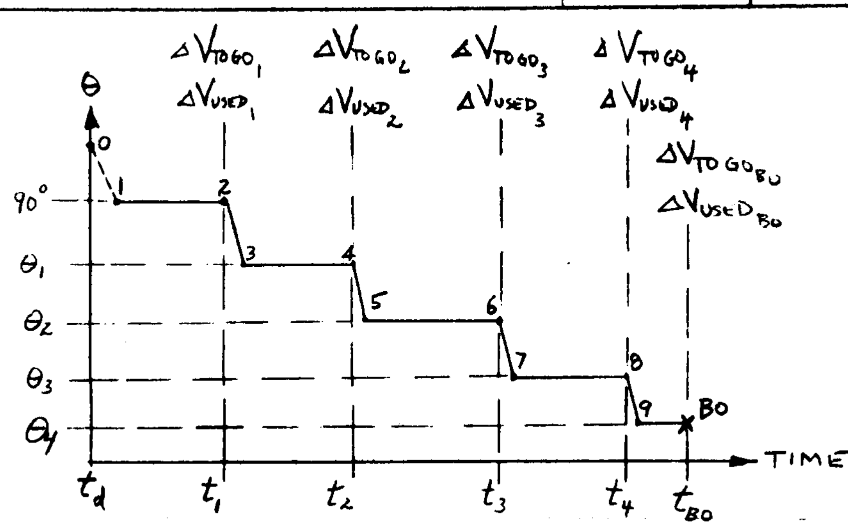
****SEE NEXT PAGE FOR DESIGNATION OF CRITICAL POINTS.

[redacted]

TABLE 3 (CONTINUED)
FIRM INERTIAL PITCH PROGRAM
ABORT PROCEDURE AND TRAJECTORY DATA

ASCENT ENGINE ONLY

ABORT POINT	1	2	3	4	5
C. TRAJECTORY PARAMETERS:					
(1) AT BURNOUT:					
h_{BO}, FT.	100,000	75,000	75,000	50,000	50,000
V_{BO}, FT./SEC.	5457.695	5469.483	5469.483	5481.328	5481.328
γ_{BO}, DEG.	0.0	0.0	0.0	0.0	0.0
$\sigma_{B_{BO}}$, DEG.	-1.4716	4.7521	7.9021	7.7745	7.7555
$\sigma_{R_{BO}}$, DEG.	8.7175	5.7220	-3.0785	-9.2675	-15.0523
(2) AT INITIATION OF TRANSFER ORBIT INSERTION THRUST:					
h_T, FT.	100,000	75,000	75,000	50,000	50,000
V_T, FT./SEC.	5457.695	5469.483	5469.483	5481.328	5481.328
γ_T, DEG.	0.0	0.0	0.0	0.0	0.0
σ_{B_T}, DEG.	32.4863	12.2072	105.2814	25.7637	81.4842
σ_{R_T}, DEG.	11.8456	6.4525	6.4634	-7.3995	-7.3963
(3) AT TRANSFER ORBIT INSERTION:					
h_i, FT.	99992.7	74994.9	74,996.3	49998.6	49998.8
V_i, FT./SEC.	5647.685	5642.979	5642.978	5591.326	5591.326
γ_i, DEG.	.0008	-.0002	-.0016	-.0015	-.0005
σ_{B_i}, DEG.	33.1390	12.7395	105.6936	26.0156	81.7351
σ_{R_i}, DEG.	11.9158	6.5121	6.5096	-7.3711	-7.3680
(4) AT APOCYNTHION OF TRANSFER ORBIT:					
h_a, FT.	134,592	870,671	870,671	535,865	535,865
(5) AT RENDEZVOUS:					
t_R, SEC.	6640.7519	6043.5534	7943.6773	5325.0285	6463.3138
σ_{B_R}, DEG.	306.3815	277.1586	370.1354	242.0054	297.7013
(6) DURING TRANSFER:					
$\sigma_{B_R} - \sigma_{B_i}$, DEG.	273.2425	264.4491	264.4418	215.9898	215.9662



DESIGNATION OF CRITICAL POINTS FOR PITCH PROGRAM

TABLE 4
FIRM INERTIAL PITCH PROGRAM
ABORT PROCEDURE AND TRAJECTORY DATA

DESCENT /ASCENT ENGINE

ABORT POINT	1	2	3	4	5
A. ABORT PROCEDURE TO BURNOUT:					
(1) AT ABORT (POINT 0 *):					
t_d, SEC	40	160	280	390.4	510.47
θ_d, DEG	187.8624	176.7033	165.5671	130.1212	90.1230
ΔV_{USED_d}, FT/SEC	541.2991	2371.8270	4623.5296	6003.8579	6661.8352
$\Delta V_{TO\ GO_d}$, FT/SEC	1848.948	3030.1710	2776.4784	1376.1421	733.164
(2) AFTER ERECTION (POINT 1):					
θ, DEG.	90	90	90	90	90
(3) AT POINT 2:					
t_1, SEC	97.8053	200.7048	329.6401	398.2015	517.2039
ΔV_{USED_1}, FT/SEC	1379.4794	3081.7950	5358.0855	6112.5625	6761.8430
$\Delta V_{TO\ GO_1}$, FT/SEC.	1010.7677	2320.0030	2041.9145	1287.4375	638.1570
(4) AT POINT 3:					
θ_1, DEG.	-66.4986	1.1296	23.8605	27.0153	26.8701
(5) AT POINT 4:					
t_2, SEC	140.0845	243.0527	332.3936	429.1212	613.8508
ΔV_{USED_2}, FT/SEC.	2041.2991	3871.8270	6123.5296	7503.8579	8161.8352
$\Delta V_{TO\ GO_2}$, FT/SEC	348.7480	1530.1710	1276.4704	4313.7881	4361.0188
(6) AT POINT 5:					
θ_2, DEG	-78.4986	-10.8704	11.3605	15.0153	14.3701
(7) AT POINT 6:					
t_3, SEC	—	326.8332	484.4255	596.1277	711.1331
ΔV_{USED_3}, FT/SEC.	—	5371.8270	7623.5296	9003.8579	9661.8352
$\Delta V_{TO\ GO_3}$, FT/SEC	—	30.1710	2403.3784	2813.7881	2861.0188
(8) AT POINT 7:					
θ_3, DEG.	—	-22.8704	-.1395	3.0153	2.8708
(9) AT POINT 8:					
t_4, SEC.	—	—	587.2138	685.3915	794.7661
ΔV_{USED_4}, FT/SEC.	—	—	9123.5296	10,503.8579	11,161.8352
$\Delta V_{TO\ GO_4}$, FT/SEC.	—	—	903.3784	1313.7881	1361.0188
(10) AT POINT 9:					
θ_4, DEG	—	—	-11.1395	-7.9847	-8.1292
(11) AT BURNOUT:					
t_{BO}, SEC.	161.2500	325.1500	641.9800	753.3500	860.3500
$\Delta V_{USED\ BO}$, FT/SEC	2390.2471	5401.9980	10,026.9080	11,817.6460	12,522.8540
$\Delta V_{TO\ GO_{BO}}$, FT/SEC.	0	0	0	0	0
B. ABORT PROCEDURE STAGING & THRUST DATA:					
(1) STAGING DATA:					
ΔV_{USED} AT STAGING, ΔV_{USED_S}, FT/SEC	NO STAGING	NO STAGING	7400	7400	7400
$\Delta V_{TO\ GO}$ BEFORE STAGING, FT/SEC	—	—	0	0	0
STAGING TIME, t_s, SEC.	—	—	467.7240	484.3141	558.4578
$\Delta V_{TO\ GO}$ AFTER STAGING, $\Delta V_{TO\ GO_S}$, FT/SEC	—	—	2626.9080	4417.646	5122.8540
(2) THRUST DATA:					
THRUST AT ABORT, T_d, LB.	10,500	10,500	10,500	6000	6000
TIME TO CUT THRUST, t_c, SEC	NO CUTBACK	297.6659	297.5310	390.4	510.47
ΔV_{USED} AT THRUST CUTBACK, FT/SEC	—	5000	5000	** 6003.8579	** 6661.8352
VALUE OF CUTBACK THRUST, T_c, LB	—	6000	6000	6000	6000

* SEE NEXT PAGE FOR DESIGNATION OF CRITICAL POINTS.

** THRUST IS REDUCED TO 6000[redacted]
AT ABORT.

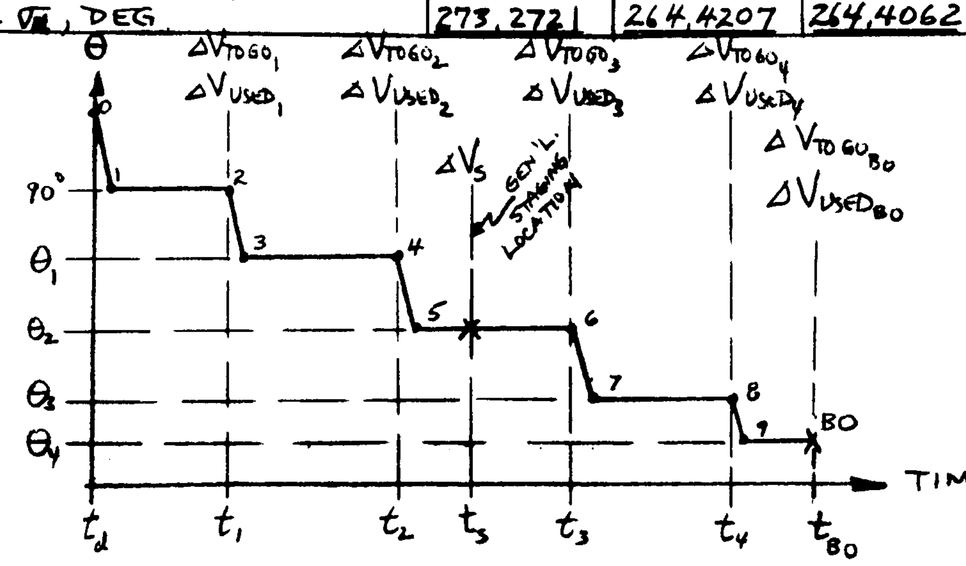
PAGE 9.
LMO 500-101
11 OCT 1963

[redacted]

TABLE 4 (CONTINUED)
FIRM INERTIAL PITCH PROGRAM
ABORT PROCEDURE AND TRAJECTORY DATA

DESCENT / ASCENT ENGINE

ABORT POINT	1	2	3	4	5
C. ABORT PROCEDURE FOR TRANSFER ORBIT INSERTION:					
(1) AT INITIATION OF THRUST:					
t_T, SEC.	312.1130	387.3217	2457.4749	1108.3777	2232.9512
θ_T, DEG.	-33.9759	-7.9248	-108.9427	-28.7572	-83.7110
ρ_T, FT.	1,275,624	786,457	788,572	884,940	884,624
ΔV_{USED_T}, FT/SEC	2370.2471	5401.9980	10,026.9080	11,817.6460	12,522.8540
$\Delta V_{TO\ GO_T}$, FT/SEC.	190.000	173.50	173.50	110.000	110.000
THRUST, T_T, LB	10,500.	6000.	3500.	3500.	3500.
(2) AT TRANSFER ORBIT INSERTION:					
t_i, SEC.	823.3230	400.5217	2467.4349	1114.1577	2237.8712
ΔV_{USED_i}, FT/SEC.	2480.2471	5575.4980	10,199.4080	11,927.6460	12,632.8540
$\Delta V_{TO\ GO_i}$, FT/SEC.	0	0	0	0	0
THRUST, T_i, LB	10,500.	6000.	3500.	3500.	3500.
D. TRAJECTORY PARAMETERS:					
(1) AT BURNOUT:					
h_{BO}, FT.	100,000	75,000	75,000	50,000	50,000
V_{BO}, FT./SEC.	5457.695	5469.483	5469.483	5481.328	5481.328
γ_{BO}, DEG.	0	0	0	0	0
$\sigma_{B_{BO}}$, DEG.	-2.0521	3.6693	9.6612	8.8956	8.3679
$\sigma_{R_{BO}}$, DEG.	8.6154	6.1190	-3.1979	-9.4158	-15.1787
(2) AT INITIATION OF TRANSFER ORBIT INSERTION THRUST:					
h_T, FT.	100,000	75,000	75,000	50,000	50,000
V_T, FT/SEC.	5457.695	5469.483	5469.483	5481.328	5481.328
γ_T, DEG.	0	0	0	0	0
σ_{B_T}, DEG.	33.0260	6.8248	108.1427	28.3072	83.3110
σ_{R_T}, DEG.	11.8470	6.4282	6.4520	-7.4001	-7.3966
(3) AT TRANSFER ORBIT INSERTION:					
h_i, FT	99,992.5	74,991.0	74,995.0	49,998.7	49,998.8
V_i, FT./SEC.	5647.689	5642.973	5642.979	5591.327	5591.327
γ_i, DEG	-.0011	-.0005	.0005	-.0003	.0001
σ_{B_i}, DEG	33.6409	7.5519	108.6913	28.5984	83.5821
σ_{R_i}, DEG.	11.9132	6.5096	6.5134	-7.3673	-7.3661
(4) AT APOCYNTHION OF TRANSFER ORBIT:					
h_a, FT	984,664.	870,847	870,847	535,865	535,865
(5) AT RENDEZVOUS:					
t_R, SEC.	6651.7030	6117.5617	8004.2589	5377.2357	6500.7632
σ_{B_R}, DEG.	306.9130	271.9726	373.0975	244.5602	299.5338
(6) DURING TRANSFER:					
$\sigma_{B_R} - \sigma_{B_i}$, DEG	273.2721	264.4207	264.4062	215.9618	215.9517

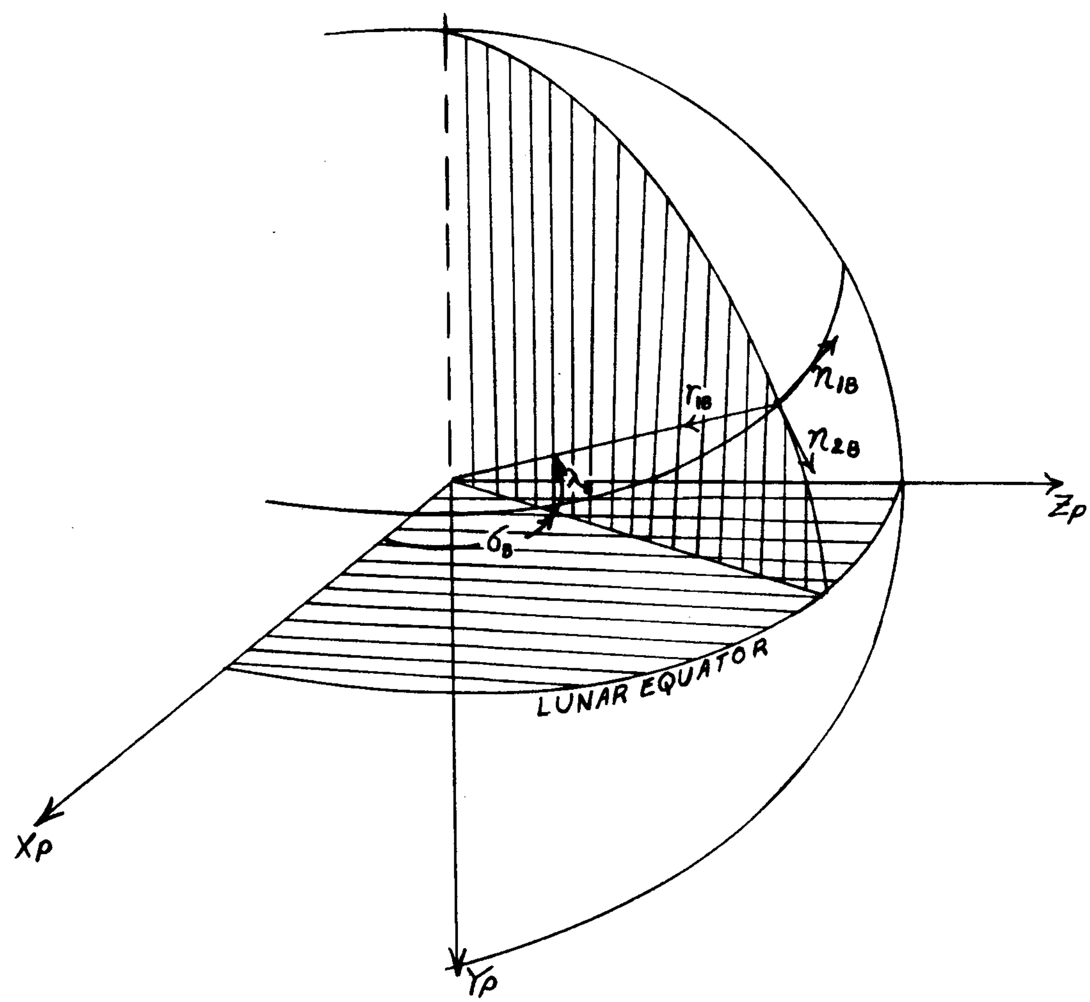


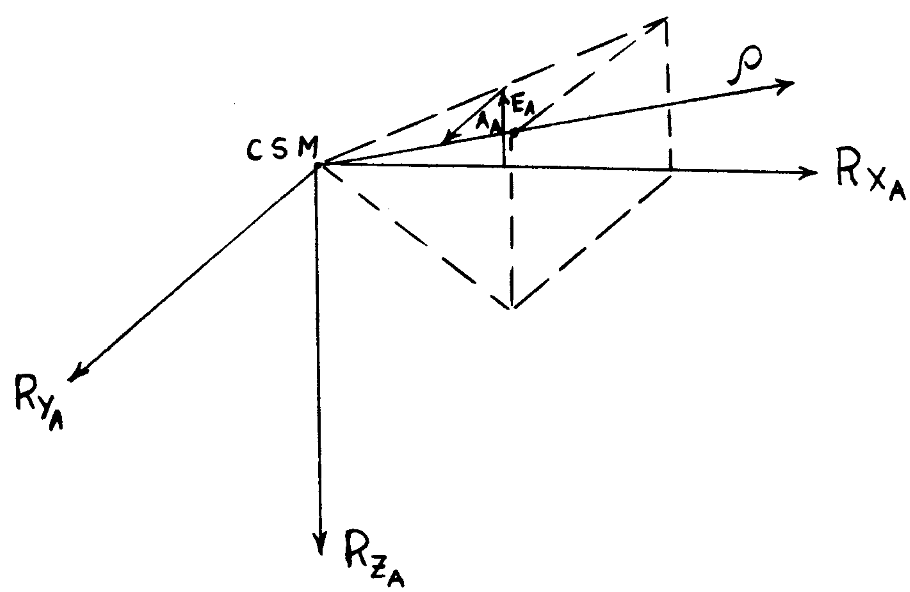
DESIGNATION OF CRITICAL POINTS FOR PITCH PROGRAM

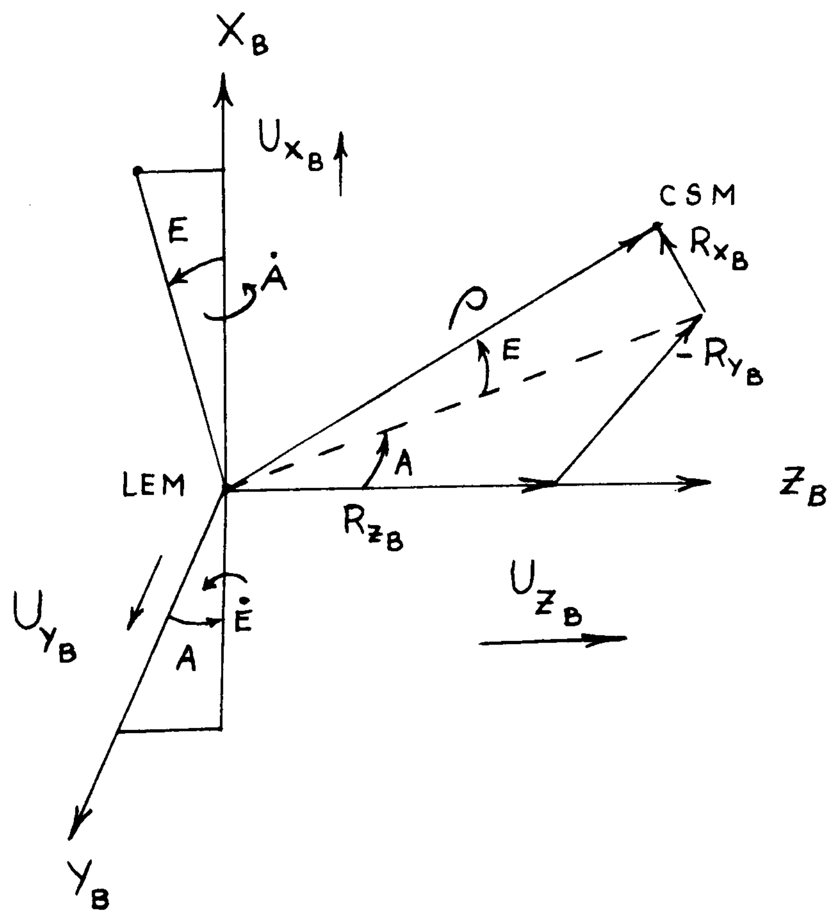
[redacted]

APPENDIX E

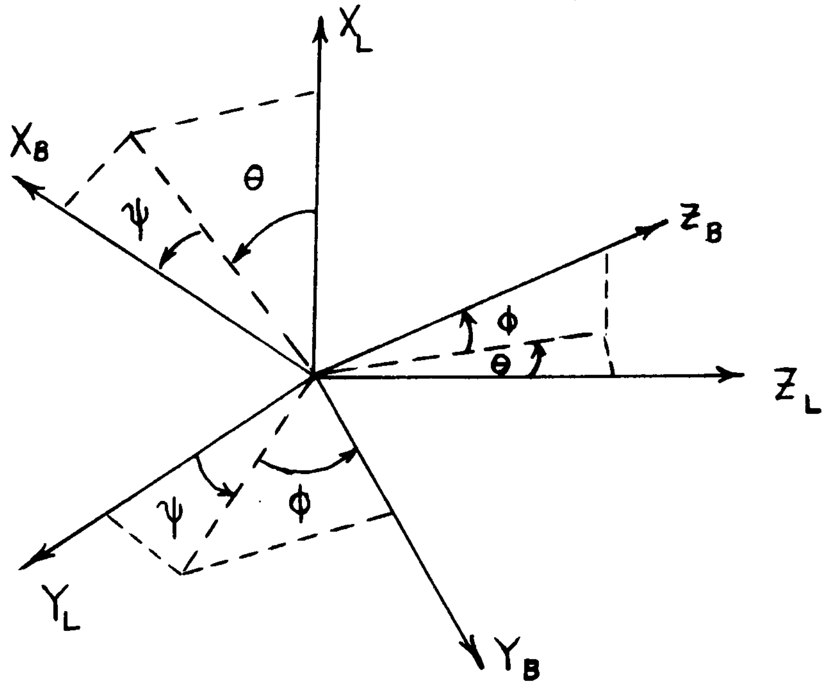
DIAGRAMS



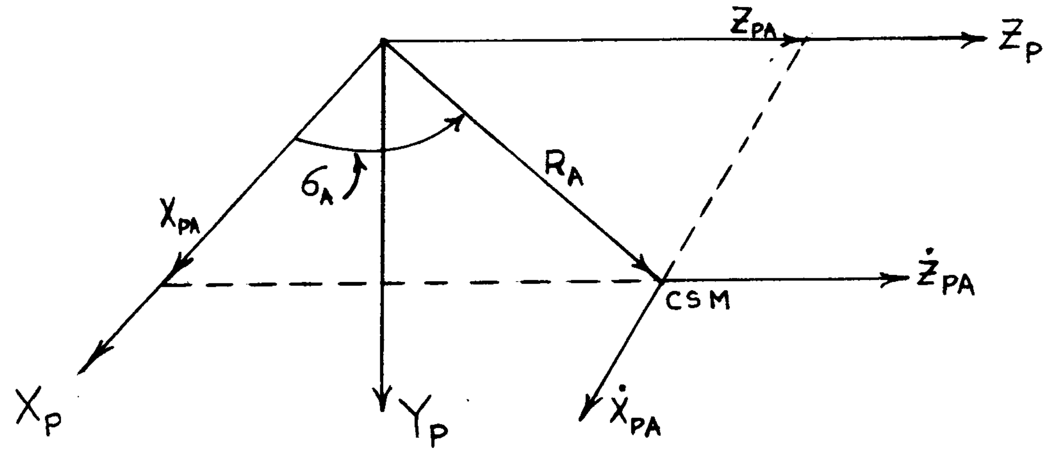

CSM
ρ
A_A
E_A
R_{X_A}
R_{Y_A}
R_{Z_A}



X_B
U_{X_B}
E
$\dot{A}$
CSM
R_{X_B}
ρ
E
R_{Y_B}
A
LEM
Z_B
R_{Z_B}
U_{Z_B}
$\dot{E}$
U_{Y_B}
A
Y_B



X_L
X_B
θ
ψ
Z_B
ϕ
θ
Z_L
ψ
ϕ
Y_L
Y_B



$$\sin \sigma_A = \frac{Z_{PA}}{R_A}$$

$$\cos \sigma_A = \frac{X_{PA}}{R_A}$$

APPENDIX F

CONTROL SYSTEM SCHEMATIC AND JET SELECT LOGIC

GAEC LMO-500-79 forms the content of this Appendix and contains the Control System Schematic and Jet Select Logic equations.

[redacted]

GROUP 4 DOCUMENT
DOWNGRADED AT 3-YEAR INTERVALS
DECLASSIFIED AFTER 12 YEARS

GRUMMAN AIRCRAFT ENGINEERING CORPORATION

LEM ENGINEERING MEMORANDUM

LMO-500-79

2 August 1963
695

From: P. Kelly, Dynamics - Systems Engineering, x1331

To: LEM Distribution Lists "M" & "N"

E. Baird	C. W. Rathke/T. J. Kelly
R. Beadle (3)	A. Whitaker
J. Kmiecik	C. Witt
R. S. Mullaney	R. Wittman
R. Peters	H. Wolf

Subject: ATTITUDE CONTROL SYSTEM FOR LEM ABORT SIMULATOR AT LTV

References: (1) LMO-500-69, "Equations of Motion for the LEM Abort Simulator", dated 17 July 1963

(2) LMO-570-75, "Work Statement for Manual Abort Study Simulation, Chance Vought Corporation, Dallas, Texas," dated 13 June 1963

Introduction:

This memo provides a description of the dynamic model of the attitude control system to be used in the LEM Abort Simulator. The control system specified herein will provide simulation of the dynamic characteristics of the flight control system with sufficient accuracy and detail to completely satisfy all primary objectives (Reference (2)) for the abort simulation. Certain simplifications in the control system dynamics have been made to reduce analog computer requirements but control system performance to pilot commands has not been compromised. The configuration of this control system reflects the contributions from and coordination with the Flight Controls Sub-system Group.

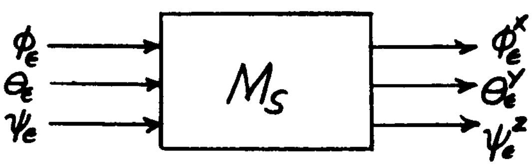
Discussion:

A schematic diagram of the attitude control system is given in Figure 1. This diagram presents the three axis control system and shows interface with pilot-operated switches, pilot controllers, automatic guidance steering signals, and vehicle body dynamics equations (Reference (1)). The operating modes for the attitude control system are listed in Table 1, and the range of parameters in Table 3.

The operating modes are selected manually by the pilot. The Attitude Mode Switch (Figure 1) selects simultaneously for all three axes, the rotational modes I,A and I,B,1,a and I,B,1,b which are shown in Table 1. The emergency modes, I,B,2,a and I,B,2,b of Table 1, are manually selected separately for each axis with the three Emergency Switches (Figure 1). The two translation control modes (modes IIA and IIB of Table 1) and the X-axis response level are manually selected with the Translation Mode Select Switch (Figure 1).

[redacted]

When the control system is put in the automatic mode (mode IA of Table 1), it operates without attitude feedback. Therefore, the guidance steering equations must form the error angles in its own coordinate system between vehicle attitude and desired attitude, which are designated herein as ϕ_ϵ, θ_ϵ, ψ_ϵ. In order for the control system to properly respond to these guidance steering errors (ϕ_ϵ, θ_ϵ, ψ_ϵ), a transformation matrix is required to transform the errors from guidance coordinates into body-axis, autopilot command signals. This matrix has been designated schematically as M_S in Figure 1. (Reference 1)



The logic equations which state the conditions which cause each of the 16 reaction jets to fire in response to error signals in the flight control system are given in Table 2. These equations will be mechanized by solid state switching networks for the simulator, referred to as the "logic box". The input to the logic equations is comprised of pitch, yaw and roll rotation torque command signals; X, Y and Z translation force commands; a pilot executed "high" or "low" X-axis translation force level selection, and pilot activated Jet Failure logic switches (located in the cockpit) for each of the four jet clusters, or quadrants. Additional switches will be supplied on the "logic box" itself for the computer operator to cause an apparent "fail-on" or "fail-off" for each of the 16 jets. The output consists of 16 on-off, constant amplitude, signals which represent the thrust command to each of the 16 reaction jets. Each individual output signal will be passed through a linear first-order lag filter, the output of which comprises the thrust of the reaction jets.

The Jet Failure Logic Switches are operated by the pilot and simulate the dual function of "RCS propellant isolation" and failure logic command. These switches will automatically change logic in the "logic box".

A detailed discussion of the functioning of the logic equations will be the subject of a separate memo. In a nominal mode, the logic provides optimum control torques and translation forces in response to any combination of simultaneous rotation and translation commands. In addition, for large rotation error signals above level, e, (Figure 1) it produces a four-jet rotation couple (smaller error response produces two-jet couples) provided that the rotation command exists in only one axis at a time. It provides for either 200 lbs. or 400 lbs. translation force along the X-axis (selectable by switch) if Y and Z axis rotation commands are not simultaneously present.

The logic operating in the failure modes produces rotation torques and translation forces which are the maximum attainable (optimum) for each particular jet failure, except for a few specific combinations of polarity of simultaneous control commands. Under these particular command combinations, the logic provides reduced and equal response for each polarity of rotation. An optimum logic would produce unequal response for opposite polarity of these command combinations. There is no obvious advantage to unequal response, (and it may even be undesirable) and the logic equations can be greatly simplified by restricting them to equal response, thereby gaining an increase in autopilot reliability. Furthermore, the pilot's task in reacting to jet failures should be simpler with four instead of eight failure switches to manipulate. Vehicle control quality is expected to be satisfactory.

The rotation torque commands consist of a train of constant amplitude pulses of varying width and frequency of occurrence, which are produced by three linear pulse ratio modulators, one for each control channel. Emergency modes of control system operation will be simulated by replacing the ratio modulator by either of the following: a fixed pulse train modulator; a direct on-off signal (constant amplitude) from pilot's controller. During control system automatic attitude hold operation, the system will limit cycle and the modulator will produce a typical pulse of 6 milliseconds duration approximately every 2.5 seconds. Limit cycle amplitude is critically dependent upon accurate simulation of the 6 millisecond pulse.

Because of the high frequency content of the thrust of the reaction jets, it is anticipated that simulation of the vehicle body dynamics in acceleration (angular) and angular velocity (p, q, r) will be done on the Analog Computer. The control system including rate gyro feedback (p, q, r), attitude error signals, modulators, select logic, jet thrust, and pilot controller signals will also be mechanized on the Analog Computer, with extra "hardware" (such as modulator, logic box, controllers, etc.) used as needed.

The translation force commands are received directly from the pilot's translation controller as on-off signals, or from a fixed pulse train modulator inserted in series with the pilot's controller.

The pulse modulators will be physically contained in the overall modulator logic box. The three rotation modulators operate in the following modes which are described below:

<u>Pulse Ratio Modulation (all nominal Control System modes)</u>

Pulse Ratio Modulation (PRM) is a pulse modulation technique whereby the pulse width (T_W) and repetition frequency (f_p) are dependent upon the normalized input signal (X). The static characteristics are defined by the following:

$$T_W = \frac{T_{W_{Min}}}{1 - X} \qquad (1)$$

$$f_p = \frac{1}{T_{W_{Min}}} \; X(1 - X) \qquad (2)$$

Where $T_{W_{Min}}$ is the minimum pulse width

The static PRM characteristics are useful only for DC or very slowly varying signals. However, if a signal other than a DC or slowly varying signal is of interest one can use the dynamic integral equations to determine the modulator operation. These equations are:

$$T_{W_{Min}} = \int_{0}^{T_W} (1 - X)\, dt \qquad (3)$$

$$T_{W_{Min}} = \int_{T_W}^{T_p = \frac{1}{f_p}} X\, dt \qquad (4)$$

The lower limit in Equation (4) is the value of T_W obtained from Equation (3).

Pulse Train Modulation

Activated by pilot selecting the manual, emergency, direct control system mode. The modulator output consists of a train of pulses fixed in amplitude, width, and frequency, in response to a signal(low level) output from the pilot's controller.

On-off

Activated by pilot selecting the manual, emergency, direct control system mode. The modulator output consists of a continuous, constant amplitude signal in response to a signal output from the pilot's controller.(high level).

PK/fl

TABLE 1

Control System Operating Modes

I. Rotational Control

A. Automatic

used for automatic guidance steering
all axes selectable simultaneously

B. Manual

1. Normal (selectable all axes simultaneously)

(a) attitude hold (rate command plus attitude hold)

(b) rate command (no attitude hold)

2. Emergency Direct

(a) Direct: Pulse Train

(b) Direct: On-off

II. Translational Control

A. Manual On-off

B. Manual Pulse Train

TABLE 2

REACTION JET SELECT LOGIC FOR LEM ABORT SIMULATION

$$T_1 = A_1\left\{Q_2(R'_2X'_1A_2 + X_2A_3) + R_1\left[(Q'_1X'_1 + Q_2A_2)A_4 + X_2A_3\right] + X_2Q'_1R'_2(C + A'_2 + A'_4)\right\}$$

$$T_2 = A_1\left\{X_1A_3(Q'_2R'_1 + Q_1) + R_2\left[X'_2Q'_2(D_r + A'_2 + A'_3) + Q_1A_2A_4 + X_1A_3\right] + Q_1R'_1X'_2(D_q + A'_4 + A'_3)\right\}$$

$$T_3 = A_1\left\{Z_2(A_3 + Y'_1\,\text{[illegible]}') + P_2Z'_1\left[(Y_1 + Y_2) + D_p + A'_2 + A'_4\right]\right\}$$

$$T_4 = A_1\left\{A_3(P_1Y'_1 + Y_2P'_2) + Y_2P_2Z'_1Z'_2 + A'_3\left[P_1Y'_1(Z_1 + Z_2) + Y_2Z'_1P'_2\right] + P_1Y_1(Z_1A'_2 + Z_2A'_4)\right\}$$

$$T_5 = A_2\left\{X_2A_4(Q'_2R'_2 + Q_1) + R_1\left[X'_1Q'_2(D_r + A'_1 + A'_4) + Q_1A_1A_3 + X_2A_4\right] + Q_1R'_2X'_1(D_q + A'_3 + A'_4)\right\}$$

$$T_6 = A_2\left\{Q_2(R'_1X'_2A_1 + X_1A_4) + R_2\left[A_3(Q'_1X'_2 + Q_2A_1) + X_1A_4\right] + X_1Q'_1R'_1(C + A'_1 + A'_3)\right\}$$

$$T_7 = A_2\left\{Z_1(A_4 + Y'_1\,\text{[illegible]}') + P_1Z'_2\left[Y_1 + Y_2 + D_p + A'_1 + A'_3\right]\right\}$$

$$T_8 = A_2\left\{A_4(P_2Y'_1 + Y_2P'_1) + Y_2P_1Z'_1Z'_2 + A'_4\left[P_2Y'_1(Z_1 + Z_2) + Y_2Z'_2P'_1\right] + P_2Y_1(Z_1A'_3 + Z_2A'_1)\right\}$$

$$T_9 = A_3\left\{Q_1(R'_1X'_1A_4 + X_2A_1) + R_2\left[A_2(Q'_2X'_1 + Q_1A_4) + X_2A_1\right] + X_2Q'_2R'_1(C + A'_2 + A'_4)\right\}$$

$$T_{10} = A_3\left\{X_1A_1(Q'_1R'_2 + Q_2) + R_1\left[Q'_1X'_2(D_r + A'_1 + A'_4) + Q_2A_2A_4 + X_1A_1\right] + Q_2X'_2R'_2(D_q + A'_1 + A'_2)\right\}$$

$$T_{11} = A_3\left\{Z_1(A_1 + Y'_2\,\text{[illegible]}') + P_2Z'_2(Y_1 + Y_2 + D_p + A'_2 + A'_4)\right\}$$

$$T_{12} = A_3\left\{A_1(P_1Y'_2 + Y_1P'_2) + Y_1\,P_2Z'_1Z'_2 + A'_1\left[P_1Y'_2(Z_1 + Z_2) + Y_1Z'_2P'_2\right] + P_1Y_2(Z_1A'_2 + Z_2A'_4)\right\}$$

$$T_{13} = A_4\left\{X_2A_2(Q'_1R'_1 + Q_2) + R_2\left[X'_1Q'_1(D_r + A'_3 + A'_2) + A_1A_3Q_2 + A_2X_2\right] + Q_2R'_1X'_1(D_q + A'_1 + A'_2)\right\}$$

$$T_{14} = A_4\left\{Q_1(R'_2X'_2A_3 + X_1A_2) + R_1\left[A_1(Q'_2X'_2 + Q_1A_3) + X_1A_2\right] + X_1Q'_2R'_2(C + A'_1 + A'_3)\right\} .$$

TABLE 2 (Cont'd)

REACTION JET SELECT LOGIC FOR LEM ABORT SIMULATION

$$T_{15} = A_4 \left\{ Z_2(A_2 + Y_2') + P_1 Z_1'(Y_1 + Y_2 + D_p + A_1' + A_3') \right\}$$

$$T_{16} = A_4 \left\{ A_2(P_2 Y_2' + Y_1 P_1') + Y_1 P_1 Z_1' Z_2' + A_2' \left[P_2 Y_2'(Z_1 + Z_2) + Y_1 Z_1' P_1' \right] + P_2 Y_2(Z_1 A_3' + Z_2 A_1') \right\}$$

Where:

T_j = Thrust Command for j^{th} jet $\quad j = 1, 2 \text{ --- } 16$

$X = 1, Y = 1, Z = 1$: Translation force command along X, Y, Z axes, respectively.

$P = 1, Q = 1, R = 1$: Rotation Torque Command about X, Y, Z axes, respectively.

Subscript 1, 2: Denotes positive rotation or translation, and negative rotation or translation, respectively.

Superscript: Primed variable indicates a <u>not</u> condition; for example $X' = 1$ when $X = 0$. All variables have value of $\overline{1}$ or 0.

$C = 1$: Enable 400 lbs. along X; $C = 0$ Enable 200 lbs.

$D_p = 1$: $|\mathcal{E}_p| >$ Threshold e_p (4 jets for rotation)

$D_p = 0$: $|\mathcal{E}_p| <$ Threshold e_p (2 jets for rotation)

$D_q = 1$: $|\mathcal{E}_q| >$ Threshold e_q $\quad D_q = 0$ $|\mathcal{E}_q| <$ Threshold e_q

$D_r = 1$: $|\mathcal{E}_r| >$ Threshold e_r $\quad D_r = 0$ $|\mathcal{E}_r| <$ Threshold e_r

$A_i = 1$: Enable jets in system A and system B in each Quad. i; where subscript, i, identifies Quad. I, II, III, IV

$A_i = 0$: Jet failure command for Quad. i. The $A_i = 0$ command can be given either because a jet sticks <u>on</u>, or because some other failure in the jet necessitates propellant shutoff to system $\overline{A}$ or B.

$a + b = a$ <u>or</u> b

$a \times b = a$ <u>and</u> b

TABLE 3

ABORT SIMULATION

CONTROL SYSTEM BLOCK DIAGRAM PARAMETERS

Parameter	Range	Units
K_R	0.1 — 1.5	Seconds
d	0.1 — 5.0	Degrees
K_M	100 — 800	Volts/Degree
e	100 — 500	Volts
g	.02 — .20	Degrees
K_g	0.1 — 0.5	Degrees/Second
j	$\frac{1}{10}$ — $\frac{1}{100}$	Seconds
T	0 — 100	Pounds

Note: $30 > D_{p,\ q,\ r} > 10$ Volts

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LEM DYNAMICS

CONTROL SYSTEM BLOCK DIAGRAM; ABORT SIMULATION

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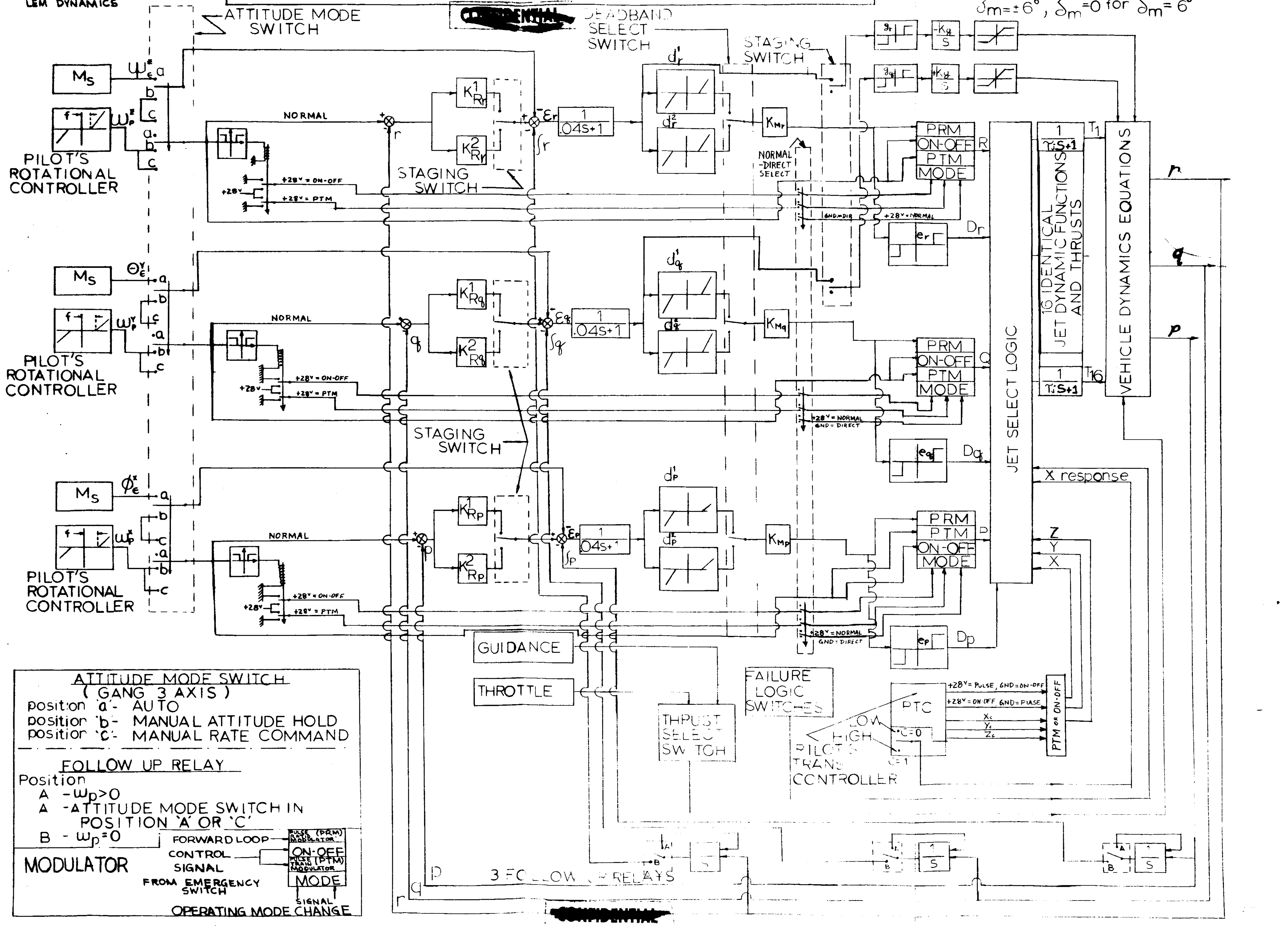

ATTITUDE MODE SWITCH
(GANG 3 AXIS)
position 'a'- AUTO
position 'b'- MANUAL ATTITUDE HOLD
position 'c'- MANUAL RATE COMMAND

FOLLOW UP RELAY
Position
A $-\omega_p > 0$
A -ATTITUDE MODE SWITCH IN POSITION 'A' OR 'C'
B $-\omega_p = 0$

MODULATOR
FORWARD LOOP CONTROL SIGNAL
PULSE (PRM) RATIO MODULATOR
ON-OFF
PULSE (PTM) TRAIN MODULATOR
MODE
FROM EMERGENCY SWITCH
SIGNAL
OPERATING MODE CHANGE

[redacted]

APPENDIX G

ABORT - TO - BURNOUT

POST TRAINING RUN SCHEDULE

TABLE 1

(INERTIAL PITCH PROGRAM)

RUN NO.	ENGINE CONFIGURATION	ATTITUDE DISPLAY	ABORT POINT
1	Ascent only	Error needles	2
2	Descent/Ascent	Error needles	2
3	Ascent only	Error needles	2
4	Descent/Ascent	Error needles	2

INITIAL CONDITIONS FOR MIDCOURSE

TRAINING RUNS

TABLE 2

PARAMETER	VALUE
X_p	5.7520362×10^6 ft.
Y_p	0
Z_p	0
X_p	0
Y_p	- 194.87252 ft. / sec.
Z_p	5580.5417 ft. / sec.
X_{PA}	6.1045873×10^6 ft.
Z_{PA}	1.0138111×10^6 ft.
X_{PA}	- 865.77876 ft. / sec.
Z_{PA}	5213.2216 ft. / sec.
m_T	114.9 slugs
X_{CG}	- 0.0332 ft. (where RCS plane equals 0 ft.)
Y_{CG}	0.0166 ft.
Z_{CG}	- 0.243 ft.
I_x	2643 slug- ft.^{2}
I_y	2283 slug- ft.^{2}
I_z	1477 slug- ft.^{2}
I_{xy}	- 11 slug- ft.^{2}
I_{yz}	- 84 slug- ft.^{2}
I_{xz}	229 slug- ft.^{2}

ATTITUDE DISPLAY INSTRUMENTATION

PRE-TEST RUN SCHEDULE

(INERTIAL PITCH PROGRAM)

TABLE 3

RUN NO.	ATTITUDE DISPLAY	ENGINE CONFIGURATION
1	E.B.[a]	D/A[c]
2	P.A.[b]	A[d]
3	P.A.	D/A
4	E.B.	A

a = Eight-ball only

b = Precision angle readouts in 3 axes and eight-ball

c = Descent/Ascent engine combination

d = Ascent engine only

CONTROL DEADBAND

PRE-TEST RUN SCHEDULE

(INERTIAL PITCH PROGRAM)

TABLE 4

RUN NO.	DEADBAND	ENGINE CONFIGURATION
1	0.5°	A
2	0.05°	A
3	1.0°	D/A
4	0.05°	D/A
5	1.0°	A
6	0.5°	D/A

ABORT - TO - BURNOUT

POST-TRAINING RUN SCHEDULE

(LINE-OF-SIGHT PITCH PROGRAM)

TABLE 5

RUN NO.	ENGINE CONFIGURATION	ATTITUDE DISPLAY	ABORT POINT
1	Ascent only	Error needles	2
2	Descent/Ascent	Error needles	2
3	Ascent only	Error needles	2
4	Descent/Ascent	Error needles	2

RUN SCHEDULE FOR ABORT-TO-BURNOUT

TEST TRIALS

TABLE 6

RUN NO.	ABORT POINT	ATTITUDE DISPLAY	ENGINE CONFIGURATION
1	1N	P.A.	D/A
2	5N	P.A.	D/A
3	3N	E.B.	A
4	1N	E.N.	A
5	3N	E.N.	A
6	30.N.	P.A.	D/A
7	5N	E.N.	D/A
8	5N	P.A.	A
9	30.N.	E.N.	A
10	3N	E.B.	D/A
11	30.N	P.A.	A
12	3N	P.A.	A
13	5N	E.B.	D/A
14	30.N.	E.B.	A
15	1.N.	E.N.	D/A
16	1N	E.B.	D/A
17	5N	E.N.	A
18	1N	P.A.	A
19	3N	E.N.	D/A
20	5N	E.B.	A
21	30.N.	E.B.	D/A
22	3N	P.A.	D/A
23	1N	E.B.	A
24	30.N.	E.N.	D/A
25	30.N.	E.N.	D/A
26	1N	E.B.	A
27	3N	P.A.	D/A
28	30.N.	E.B.	D/A

TABLE 6 (CONT.)

RUN NO.	ABORT POINT	ATTITUDE DISPLAY	ENGINE CONFIGURATION
29	5N	E.B.	A
30	3N	E.N.	D/A
31	1N	P.A.	A
32	5N	E.N.	A
33	1N	E.B.	D/A
34	1N	E.N.	D/A
35	30.N.	E.B.	A
36	5N	E.B.	D/A
37	3N	P.A.	A
38	30.N.	P.A.	A
39	3N	E.B.	D/A
40	30.N.	E.N.	A
41	5N	P.A.	A
42	5N	E.N.	D/A
43	30.N.	P.A.	D/A
44	3N	E.N.	A
45	1N	E.N.	A
46	3N	E.B.	A
47	5N	P.A.	D/A
48	1N	P.A.	D/A

RUN SCHEDULE FOR MIDCOURSE

TEST TRIALS

TABLE 7

RUN NO.	ATTITUDE DISPLAY	INITIAL CONDITIONS
1	c	3
2	c	1
3	b	1
4	a	4
5	a	2
6	a	3
7	a	1
8	b	3
9	c	2
10	c	4
11	b	4
12	b	2
13	b	2
14	b	4
15	c	4
16	c	2
17	b	3
18	a	1
19	a	3
20	a	2
21	a	4
22	b	1
23	c	1
24	c	3

RUN SCHEDULE

FOR COMPLETE ABORT TRAJECTORY TEST TRIALS

TABLE 8

RUN NO.	VEHICLE CONFIGURATION	DEGRADATION CONDITION
1	A	D
2	D/A	R.F.
3	A	N
4	A	R.O.
5	A	D.P.
6	D/A	N
7	D/A	R.O.
8	D/A	C
9	A	R.F.
10	A	C
11	D/A	D.P.
12	D/A	D